AIRPLANE DESIGN MANUAL

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INTRODUCTION

This book has been written to fill what appears to the author to be a gap in aeronautical literature, an introduction to the art of airplane design, with the needs of the student, the young engineer, the draftsman and the student working on his own especially in view. While aerodynamics, stress analysis and other aspects of airplane design have been covered many times, experience in dealing with senior aeronautical students has shown that such men experience considerable difficulty in coordinating their knowledge and efforts in approaching the difficult problem of actually beginning the design of a new machine, and carrying on the work systematically. From time to time notes have been prepared for student use and these have gradually evolved into the present work.

In view of the rapid growth and complexity of the subject, it is too much to hope that the entire field has been adequately covered; still teaching experience indicates that such a manual is helpful to instructors and students alike.

It is of course expected that the student shall supplement the present text by investigations of his own, by studying the latest designs at the airport, or from descriptions in the technical press or by study of the numerous research publications published by the Government Printing Office and the great engineering societies, even though an attempt has been made to make each chapter of the book as complete in itself as possible.

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I. Procedure in Design

No task can be intelligently executed unless a definite goal has been set and a line of attack or orderly form of procedure has been adopted. There may be different ways of obtaining the same objective, but mistakes and unnecessary work will be avoided if a definite plan is made before any real work starts.

Type

It is not sufficient to say "Let's build an airplane." The question is: What kind of airplane—an open or a closed type, a sleek racing monoplane or a large flying boat? The first thing to be done is to write down a set of such definite specifications that any designer who receives them may be able to design an airplane which meets the original design proposer's intentions. The procedure is much the same as that of buying a family car. The term "family car" immediately sets one specification: the buyer knows that he is not going to get a truck or a roadster or a racing car. The price that the buyer can meet will automatically set another specification, and so it goes.

Specifications for an airplane are far more comprehensive.

Consider, for example, the type of airplane it may be. An airplane may be either a *monoplane* or a *biplane*—at least these are the two conventional types. If it is a monoplane, the wing may be unsupported externally, in which case it is known as a *full cantilever monoplane* or the wing may be externally supported either by struts or by wires, in which case it is known as a *semi-cantilever monoplane*. Moreover, the wing may be placed at the bottom of the fuselage when it is known as a *low wing monoplane;* or the wing may be placed halfway between the top and bottom of the fuselage, and the airplane is a *mid-wing monoplane;* or again, the wing may be at the top or above the fuselage in which case the airplane is known as a *high wing* or a *parasol monoplane,* respectively.

The same applies to a biplane. The two wings may not have the same areas, or the same planform, or the same airfoil. There may be large forward or positive stagger of the upper wing relative to the lower, and perhaps more dihedral for one wing than for the other. The combinations are almost infinite especially when one considers that changes may be made in structure, in materials, in planform, in stagger, in angle of incidence, in airfoil sections, in decalage,

in gap-chord ratios, in wing placement relative to the fuselage, in distribution of wing areas and a host of other variables.

The variables just noted apply only to the wing. Consider the fuselage. It may be round, oval, square, elliptical, rectangular or a combination of these cross-sections. It may be shallow or deep, it may be wide or narrow; it may have an open cockpit or an enclosed cabin; it may be constructed of almost any material and in an infinite number of ways. For each material and specific function, there is a definite, desirable shape of fuselage.

The landing gear also offers enormous latitude in design. The conventional landing gear having two wheels forward with a tail wheel rearward; or the reverse order with a front or nose wheel and two wheels slightly rearward, popularly known as the "tri-cycle" landing gear, may be employed. The landing gear may have a through axle of the type used during the war period, or a split-axle type developed later. Moreover they may be non-retractable or retractable.

These are just a few indications of what the design trend might be; moreover, it should be noticed that the flying boat has not even been mentioned.

Familiarity with different types of airplanes will help the potential designer in determining for himself what type is best suited to a specific duty. However, it is not the scope of this book to discuss such adaptation. Perhaps the best way to learn that is to read various technical aeronautical magazines and books; to study details, collect and correlate them and soon there will emerge a definite order or set of rules which one may follow.

Power Plant

The power plant will be discussed in detail in a subsequent chapter. However, a brief discussion of the place the power plant plays in the original specifications may not be amiss here. In many cases, operating companies of aircraft may specify the type and number of engines—either because of known fuel economy, or efficiency and dependability under certain operating conditions, or because of possible interchangeability with existing equipment.

An airline accustomed to maintain and operate radial air-cooled engines will be loath to use airplanes using liquid-cooled Vee engines, for example, since its personnel may not be trained or sufficiently experienced to handle the new type of engine.

The reasons for choosing a certain engine may be many and the

section on power plants should be studied before writing the specifications.

The specifications may designate a particular engine, although it is more likely that the number of engines will be designated, for it is quite possible to obtain one engine or two engines delivering the same total horse-power.

Payload and Crew

The gross weight of the airplane is largely dependent upon the requirements for payload and crew. It should be quite obvious that if a crew of three—a pilot, a co-pilot and a radio man, for example—is required, some provision must be made for it, and such provision will affect the size of the cockpit as well as the fuselage and eventually the gross weight. Likewise, provision for mail and express will be entirely different from provision for passengers.

The payload includes all load from which revenue is obtained. It includes passengers, mail, baggage and express. The crew includes pilot, co-pilot, mechanics, navigators, radio men, stewards and any other employee required for specialized work.

The larger the aircraft becomes, the larger the crew is likely to be. Some indication of the eventual size of the airplane to be designed can be gained by looking at the size of crew required. In the same way, the number of passengers carried has a direct bearing on the size of the fuselage and the gross weight: the greater the number of passengers to be carried the larger, and wider, and higher the cabin and therefore the larger the fuselage will be. Not only does the increased number of passengers itself increase the weight, but the structure will weigh more because of increased size.

Actually, the gross weight of the airplane can be estimated if the weight of the payload, crew, fuel and oil are known since an analysis of a large class of airplanes shows that there is a definite relationship between the two weights.

Military airplanes have a different type of payload, usually called fixed equipment or disposable load as the case may be. This consists of guns, ammunition, bombs and other military equipment. Special provision must be made for these; therefore these items have a definite bearing on the type as well as the weight.

It is very important to know as much as possible about the load the airplane is to carry because these are the items for which the designer has to make proper provision but over whose weight, size or location in the airplane he may have little or no control.

Performance Requirements

Unless the airplane is designed for private use, the performance requirements are set by the ultimate purchaser. It takes but little thought to discover that where competition is as keen as on American airlines, the speed of the airplane must be as high as possible in order to obtain attractive schedules. But where there is no competition, as for example in Northern Canada, a far slower airplane may be desirable. A slower airplane is more economical in the latter case because of the smaller horsepower and less fuel required to carry practically the same load.

Likewise an airplane operating over mountainous territory will need a high service ceiling in order to clear the mountains, whereas a lower service ceiling would do over low level country.

The performance required for the airplane will have a direct bearing on the number, type and horsepower of the engines, as well as the type and design of wing, fuselage, and perhaps landing gear. Perhaps the ultimate criterion of a good airplane is its performance in relation to the load carried and the conditions to be met.

Performance requirements are becoming more and more inclusive as reference to the later chapter on preliminary performance calculation will indicate. In general the performance requirements cover cruising ranges at various cruising speeds and at various altitudes; top speeds at various altitudes; landing speeds; rates of climb, and service ceiling. A reading of a Department of Commerce specification for a particular design given at the end of this chapter will also help the reader to appreciate the various items coming under the heading of performance.

Procedure in Design

All of the foregoing discussion deals with specifications, which are only one part of the plan to be considered in designing the airplane. From the moment a new design is considered until the final drawing leaves the drawing board, a definite plan is followed in evolving the design. The following procedure is customary:

1. Study of specifications
2. Study of similar types of airplanes
3. Power plant survey
4. Preliminary three-view
5. Preliminary weight estimate
6. Airfoil selection
7. Detailed weight estimate
8. Balance diagram
9. Inboard profile
10. Wing structural layout
11. Landing gear layout

12. Tail surface structural layout
13. Fuselage structure layout
14. Final three-view
15. Preliminary longitudinal stability calculations
16. Preliminary performance calculations

Very often it is desirable to carry on several steps at once since they are inter-dependent. These steps are discussed in detail in the various chapters that follow. Two detailed specifications are given below.

Operating companies may put out specifications covering every conceivable detail. For school work, the following brief specification used by the author may be of interest.

Airplane Design

The following work must be done during the first term and approved by the instructor in charge:

1. Statement as to general purpose, main features of design, objects desired, etc.
2. Weight analysis.
3. Three-view drawing.
4. Balance diagram.
5. Description of the structure of the airplane with reference to the following drawings:
    a. Wing assembly.
    b. Control system.
    c. Chassis assembly, with schematic diagram of the retraction gear.
    d. Tail surface assembly.
6. Description of the power plant, instruments and equipment with special references to drawings.
7. Description of passenger and pilot accommodations, heating and ventilating system, noise proofing provisions of cabin, etc., with reference to drawings.
8. Preliminary performance calculations.
9. Preliminary longitudinal and control calculations.

The following work must be done during the second term and approved by the instructor in charge. All stress analyses are to be in accordance with the Department of Commerce requirements.

1. Stress analysis of wing structure for at least two conditions.
2. Stress analysis of the landing gear for at least two conditions.
3. Stress analysis of fuselage for two or more conditions.

4. Stress analysis of horizontal tail surfaces and elevator control system.
5. Detail fitting design.
   a. Front spar fuselage attachment.
   b. Rudder control assembly.
   c. Tail support fittings.

Specifications

The following are the specifications for the air transport plans to be designed during the academic year:

Type: A cabin passenger plane of the most modern design, with lift increase device and retractable landing gear. Originality of design will be encouraged unless fundamental errors of judgment exist. All metal construction is encouraged.

Engine: The Pratt & Whitney, double-row radial engine, rated at 800 horsepower at 2400 r.p.m. with a propeller gear ratio of 3.2, or Wright Aeronautical Corporation engine of 735 horsepower.

Propeller: A controllable pitch propeller is required as standard equipment.

Starter: Eclipse Electric Starter.

Useful Load (Minimum): Pilot and co-pilot, 10 passengers, 400 pounds of mail or baggage, fuel and oil for 750-mile range.

Instruments: Blind flying and additional navigation instruments and power plant instruments.

Equipment: Fire extinguishing apparatus; safety belts; navigation and landing lights; heating, ventilating and lighting systems; two way radio; aircraft and engine log books; first aid supplies; independent wheel brakes; toilet facilities; soundproofing.

Performance: Landing speed not exceeding 65 miles per hour; take-off within 100 feet at sea level; climb in feet—first minute—fully loaded —650 ft. per minute or better; maximum speed, 210 miles per hour; range at 160 m.p.h., 800 miles.

To illustrate the extent of design specifications, the following specification issued a few years ago (but no longer in force), for a bid for a twin-engine, six-place cabin monoplane by the Bureau of Air Commerce, Department of Commerce, may be of interest. Only a few modifications have been made by the author.

Specification For Twin-Engine, Six-Place Cabin Monoplane

I. *General Specification*

1. This specification covers the requirements for twin-engine, six-place cabin monoplanes.

2. The current issue of the Bureau of Air Commerce, "Airworthiness Requirements for Aircraft," forms a part of this specification.

3. Prior to delivery the airplanes shall have been granted an Approved Type Certificate.

4. An adequate full scale mock-up shall be made.

5. The cabin interior shall be at least 5 ft. between floor and head lining and 5 ft. between side walls at the master section. The distance between seat backs shall be at least 30 inches. The seats shall be at least 19 inches inside width. There shall be no division between the pilot's cockpit and the passenger compartment. The seat adjacent to the pilot's seat shall be available for use for either a copilot or a passenger.

II. *Performance Requirements*

With fixed equipment as listed hereinafter and a useful load consisting of:

1 pilot	170 lb.
5 passengers	850 lb.
baggage	200 lb.
cargo	200 lb.

Fuel and oil sufficient for a cruising range of 500 miles at any altitude between sea level and 5000 feet.

the minimum performance acceptable when using fuel of not more than 80 Octane will be:

* *a.*	High speed in level flight (m.p.h.)	175
** *b.*	Landing speed with power off (m.p.h.)	65
c.	Ceiling (one engine dead) (ft.)	6000
d.	Distance from start to clear 50-ft. obstacle (ft.)	1500
e.	Distance to stop after clearing 50-ft. obstacle (ft.)	1000

* At any altitude from sea level to 5000 ft. altitude (Standard Air).
** Standard Air.

III. *Special Service Requirements*

Sufficient gasoline and oil tankage capacity shall be provided to attain the maximum possible cruising range with a useful load consisting of:

2 pilots	340 lb.
baggage	200 lb.
cargo	200 lb.

This cruising range shall not be less than 1000 miles.

IV. *Fixed Equipment*

The following equipment will be installed:

Sperry automatic pilot
Rich-Lux CO_2 fire extinguisher equipment
2 one-quart hand fire extinguishers
First aid kit
Engine covers
One set of emergency tools
One set of aircraft tie-down equipment
Approved type landing lights, fixed type
Navigation lights and red warning light
De-icing equipment adequate to permit continuous operation under icing conditions
Engine starters, hand and electric
Cabin heating and ventilation equipment
Hydraulic parking and service brakes
2 twelve to sixteen volt batteries
Controllable pitch propeller
Flares as required for night flying
Engine-driven electric generators
1 radio compass and standby receiver
2 separate radio beacon receivers
2 separate radio power units
1 dual transmitter
(Western Electric Type W 4 A or equivalent)
Antennae required for the radio equipment
1 two-quart thermos bottle

V. *Instruments*

The following instruments shall be provided and installed:

1 Pioneer airspeed indicator
1 Pioneer electrically-heated Pitot-Static airspeed head
1 Kollsman altimeter
1 Kollsman sensitive type altimeter
1 Pioneer bank and turn indicator
1 Pioneer rate of climb indicator
1 Pioneer compass with new type compensation unit
1 Pioneer clock (with sweep second hand)
1 Sperry artificial horizon
1 Weston ice warning indicator and free air thermometer
2 electric tachometers
2 Pioneer oil pressure gauges
2 Pioneer fuel pressure gauges
2 Weston oil temperature gauges
2 Weston carburetor air temperature gauges
2 Weston electric fuel quantity gauges
1 or 2 volt ammeters
2 engine temperature indicators

2 supercharger pressure gauges, if engine is supercharged
2 fuel pressure warning lights
1 flap position indicator
1 landing gear signal if wheels retractable

VI. *General Design Requirements*

1. Provision shall be made for as nearly perfect visibility for the pilot as is possible to obtain for adverse weather and night flying and to facilitate getting into restricted landing areas.

2. Special consideration shall be given to those airplane characteristics which relate to landing and take-off, ground handling, character and angle of glide, vision for landing and take-off, distance of landing and take-off roll, controllability and stability at minimum flying speeds and to all characteristics essential to safety.

3. Special attention shall be given also to those characteristics affecting maintenance costs such as reproduction costs of component parts, ease with which component parts may be inspected and repaired or replaced and probable length of life of parts.

4. In addition to meeting the requirements for an Approved Type Certificate, the airplane shall have no undesirable flying characteristics.

VII. *Detail Requirements*

A. Controls:

1. A dual set of controls readily removable shall be provided. All controls not duplicated shall be so located as to be conveniently accessible to the co-pilot but no control shall be so located as to constitute a hazard when the front seat is occupied by a passenger.

2. All control surfaces shall be adequately balanced.

3. Adequate controllable trimming tabs shall be fitted to the elevator and rudder.

4. Ball- or roller-bearing hinges shall be used throughout.

5. The rudder pedals shall be adjustable fore and aft.

6. The rudder pedals shall control the differential action of the brakes.

Note: The brake controls preferably should be mounted on the rudder pedals and provided with additional parking brake lock.

7. A suitable parking brake capable of holding the plane against maximum thrust shall be provided.

8. Full operation of high lift devices shall not require an independent readjustment of the trim control for trim.

9. Electric, hydraulic or mechanical operation of high lift devices and retractable landing gear, if provided, is required and full operation in either direction shall not require more than one minute.

B. Fuselage:

1. The pilot's and co-pilot's seats shall be adjustable vertically.

2. The pilot's windows shall be so arranged that there will be no unsatisfactory reflections.

3. Movable panels shall be provided for visibility in bad weather and provisions shall be made to deflect the air stream so that rain or snow will not enter when the panels are open.

4. An emergency exit of ample size and with rapid release shall be incorporated in the roof in the pilot's vicinity. Cockpit windows will be so arranged that the pilot can reach outside to clean them.

5. All windows shall be made of shatterproof glass free of flaws.

6. A heating and ventilating system shall be provided which can maintain a temperature of at least 70°F. in the cabin with an outside temperature of –20°F. and which can maintain a temperature inside the cabin not more than 4°F. above the outside temperature when the latter is in excess of 66°F. Adequate ventilation shall be provided under all circumstances and the air shall be clean and pure. There shall be no undue drafts created by the heating or ventilating system.

7. The cabin shall be sound-insulated to maintain a noise level below 75 decibels when cruising under normal conditions.

8. An adequate cabin lighting system shall be provided with control from the pilot's seat, inasmuch as there is no partition provided. Provision should also be made for individual lights at the seats.

9. Suitable means shall be provided for readily lifting the rear end of the fuselage with a chain fall. Means shall also be provided for readily hoisting the airplane for inspection of the landing gear.

10. A suitable baggage compartment and a suitable cargo compartment, each with ample doors, shall be provided. Provision shall be made for hold-down straps in both compartments. There shall be no sharp or rough projections which might damage cargo or baggage.

C. Wings:

1. Satisfactory provision shall be made for the internal inspection of the wings and for the inspection and servicing of all fuel, electrical and control systems inside of the wing.

2. Tie-down rings shall be provided under the wing tips.

D. Landing Gear:

1. Wheel brakes and tires of ample load and braking capacity shall be used.

2. Oleo type shock absorbers shall be used.

3. If the chassis is retractable the retracting gear shall function properly under all operating conditions found in both summer and winter. If retractable it shall be positively operated in both directions.

4. Ample provision shall be made in the design to prevent mud or ice interfering with the operation of the wheels or the retracting gear.

5. If the gear is retractable, a positive lock shall be provided holding gear in both retracted and extended position.

6. The capacity of the shock absorbers and the location of the landing gear shall be such that the airplane can be safely flown onto the ground under conditions which would be encountered in blind landing.

7. The tail wheel shall swivel 360° and shall be provided with a lock operative from the pilot's cockpit for holding the wheel in neutral position.

8. The tail wheel shall be quiet in operation and shall not shimmy during take-off, landing or taxying, whether in locked or unlocked position.

E. Engines and Engine Nacelles:

1. The engines used shall have Approved Type Certificates.

2. The engines and engine nacelles, if any, shall be quickly detachable and shall be interchangeable from right to left.

3. The engine cowling shall be readily detachable.

4. The engines shall be properly equipped to prevent icing under all conditions.

5. The exhaust collector shall be of "Toncan-iron" heat-resistant material or its equivalent. It shall be so installed that any one cylinder may be removed without removing the ring, if a ring collector is used.

6. Provision shall be made for readily using an outside source of electric power for the engine starter.

7. The engines shall be rubber mounted.

F. Fuel System:

1. The fuel tanks shall be of welded aluminum construction or its equivalent, shall be adequately supported and shall be contained in the wing. They shall be capable of withstanding the Army vibration test.

2. The fuel tank fillers shall have a 3-inch diameter opening and shall be so designed that fuel which may be spilled in filling will not get into the wing. A quickly removable and replaceable leak-proof cap shall be provided.

G. Oil System:

1. An adequate oil level measuring device shall be provided, which will also show amount of oil while in flight.

2. Adequate control of the oil temperature under all conditions of operation, in both summer and winter, shall be provided.

3. Provision shall be made for readily draining all of the oil when the airplane is on the ground, and means shall be provided for heating oil without draining.

H. Instrument Board:

1. The instrument board shall be mounted to reduce vibration within the permissible limit set by the instrument manufacturers. All blind flight instruments shall be so located as to be approximately in line with the horizon while in normal flight requiring the least movement of the pilot's eyes. An adequate lighting system shall be provided with rheostat control. Suitable provision should be made for such additional instruments as: cone marker, blind landing, etc.

2. All electrical instruments and controls with the exception of the electrical engine instruments shall be mounted on an instrument board independent of the board containing the flying and engine instruments and shall be shielded with a metal cover so hinged that the connections are readily accessible from the cockpit.

3. All horizontal position instruments shall be so placed that the hands will lie in a horizontal plane at cruising speed. All directional instruments shall be in a vertical line.

4. Fuel and oil pressure lines shall have Tite-flex or its equivalent at the instruments. All other connections to the in-

struments shall have extra-flexible connections of a serviceable type.

5. Flying instruments shall have a dual source of power, one from each engine, and each system shall be independent of the other with a gauge showing vacuum in inches of mercury.

I. Electrical Equipment:

1. All wiring shall be run in aluminum conduit; radio wiring, low voltage wiring and high voltage wiring shall be in separate conduits. The conduit systems shall be oil-tight and the conduits shall attach to the junction boxes by means of detachable joints. Junction boxes will be provided at points which will permit easy removal for installation of new wiring.

2. All terminal blocks shall be housed in aluminum boxes provided with covers held in place by wing nuts.

3. Flexible conduits shall be used only where necessary.

4. A one-wire electrical system shall be used except to the radio transmitter and instrument board lights.

5. The batteries shall be located where they may be quickly removable or serviced from the outside of the airplane.

6. A master battery switch suitably shielded shall be located not more than 12 inches from the batteries and this switch shall be operative from the pilot's cockpit.

7. The negative terminal of the batteries shall be grounded.

8. All electrical apparatus shall be suitably shielded and the airplane shall be properly bonded for radio reception.

VIII. *Mock-up Requirements*

1. The mock-up shall represent clearly the installation of all power plant, fixed equipment and useful load material. The mock-up shall include the cockpits and other portions of the airplane in the immediate vicinity of the installed material, in which places all structural members shall be represented accurately as to size and location and all pieces of cowling shall be cut to shape and proper form. The quality and kind of material used in the mock-up are not important and it shall be sufficiently strong to permit the carrying of the required installations and crew.

2. All power plant, fixed equipment, useful load material, and instruments shall be installed but if the actual material is not available dummies correct in dimensions may be used.

3. Photographs of the mock-up shall be supplied.

IX. *Inspection and Tests*

1. The manufacturer is solely responsible for compliance with these specifications, for material and workmanship, for flight tests and for proper inspection during the process of manufacture. However, any authorized representative of the Department of Commerce shall be afforded all reasonable means for making inspection if and when the Department elects to do so, but such inspections shall in no way relieve the manufacturer from responsibility. The manufacturer shall make such flight tests in addition to those herein specified as may be required by the Department of Commerce to determine whether or not the airplane was built in accordance with these specifications. A representative of the Department of Commerce may pilot the airplane during such parts of these tests as may be deemed necessary.

2. Upon completion of and prior to acceptance, the following actual performance characteristics shall have been obtained by manufacturer's test (and under the supervision of a Bureau of Air Commerce representative) and the information obtained delivered to the Bureau of Air Commerce. The procedure and methods to be followed in obtaining this test data will follow the method approved and prescribed by the Bureau of Air Commerce.

a. *Take-off performance with all engines functioning normally.*

(1) Ground run to minimum take-off speed.

Weight	*Min. Take-off Speed (m.p.h.)*	*Distance (ft.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Max. Gross				
Light				

(2) Horizontal distance required for acceleration from standing start to minimum speed required for level flight with one engine dead.

Weight	*Min. Dead Engine Speed (m.p.h.)*	*Distance (ft.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Max. Gross				
Light				

(3) Horizontal distance required for acceleration from standing start to speed of best angle of climb.

Weight	*Best Climbing Speed (m.p.h.)*	*Distance (ft.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Max. Gross				
Light				

(4) Best angle of climb.

Weight	*Angle (degrees)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*

b. *Performance in air with all engines functioning normally.*

(1) High speed in level flight (at max. gross weight).

Altitude (ft.)	*High Speed (m.p.h.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Sea Level			
5000			
Critical Altitude of Engine			

(2) Service ceiling (where best climb is 100 ft. per min.).

Weight	*Ceiling (ft.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Max. Gross at Start			
Light			

(3) Cruising speed (approx. 70% rated power—dependent upon operating instructions—max. gross weight only).

Altitude (ft.)	*Cruising Speed (m.p.h.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Sea Level			
Critical Altitude of Engine			

(4) Fuel and oil consumption at cruising speed—max. gross weight.

Fuel Consumption (gals. per hr.)	*Oil Consumption (gals. per hr.)*

(5) Minimum level flight speed.

Speed with Flaps Retracted (m.p.h.)	*Speed with Flaps Extended (m.p.h.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*

c. *Performance in air with one engine dead.*

(1) Service ceiling (where best climb is 100 ft. per min.).

Weight	*Ceiling (ft.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Max. Gross			
Light			

(2) High speed in level flight with engines at maximum power authorized for continuous operation, maximum gross weight only.

Altitude	*Speed (m.p.h.)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
(a) Sea Level			
(b) Engine Ceiling			

(3) Fuel and oil consumption for conditions 2a and 2b above.

Fuel Consumption (gals. per hr.)	*Oil Consumption (gals. per hr.)*

(4) Best angle of climb.

Weight	*Angle (degrees)*	*Engine Manifold Pressure (in.)*	*Propeller R.P.M.*
Max. Gross			
Light			

d. *Landing.*

(1) Steepest gliding angle practicable for landing approach—flaps extended, maximum gross weight—no power.

Angle (degrees)	*Speed (m.p.h.)*

(2) Length of ground roll from 3-point landing—maximum gross weight only—brakes used—landing for conditions as in d-1.

Flap Position	*Distance (ft.)*	*Landing Speed (m.p.h.)*

Factors Affecting Type of Airplane

In determining a suitable type for an airplane, the factors affecting performance are of utmost importance. These are considered below:

Take-Off. Where heavy loads are to be carried, and the take-off distance of an airport is limited, a quick take-off is desirable.

Unless an aileron type flap or a rearward extending flap is employed, it is very desirable to have a moderately low wing loading, a large span and an airfoil with a high effective lift over drag ratio.

Ceiling. To obtain a high service ceiling, the airplane should be so designed as to have:

a. A low horsepower loading (i.e. the ratio of gross weight to horsepower)
b. A low wing loading (i.e. the ratio of gross weight to wing area)
c. A large span for a given area
d. An airfoil with a high effective lift over drag ratio

High Speed. High speed may be obtained by a series of factors, briefly listed below:

a. Low horsepower loading
b. Efficient propeller
c. Very clean design—all drag reduced to an absolute minimum

Cruising Speed. The same considerations holding for high speed hold for cruising speed. However, there must be proper coordination between economical operation of the power plant and a desirable cruising speed. For any given design, this can be determined best from wind tunnel tests and full flight tests.

PROBLEMS

1. Write a possible specification around an engine developing 325 horsepower.

2. What factors affect the writing of a specification for an airplane? Hints: terrain, purpose, range, etc.

Discuss the specification for the twin-engine monoplane from point of view of:

a. comfort

b. performance compared with similar types that you may know

c. private flying appeal

d. feeder airline possibilities

II. Preliminary Weight Estimate

Unless the beginning designer is copying an existing design, the first data that he will need to begin his design after he has been given or has written his own specifications, are various estimates as to the approximate gross weight of the airplane to be designed. For on this gross weight will depend the wing area required to meet the specified landing speed. Once the wing area has been determined, the length of the fuselage and the areas of the horizontal and vertical tail surfaces are also quite accurately known, since airplanes to-day have definite relationships of areas and dimensions of fuselages, tail surfaces and even the landing gear tread, in terms of the span and area of the wing.

When the length of the fuselage is known, its weight can be estimated; likewise if the areas of the tail surfaces are known, their weight may be estimated, and so the data accumulates.

When an approximate figure for the gross weight has been decided upon, the size and weights of the landing gear may be found and so more and more data accumulates until the structural weight of the complete airplane has been determined. It may be only approximate, but a series of refinements will finally give a weight estimate —and it is called an estimate until each part has been built and weighed—that will be surprisingly accurate when the airplane is finally built.

The methods by which the weight is estimated are, on the whole, empirical and every bit of information that can be gleaned from the specifications and from existing similar designs will be helpful.

An experienced designer need only look at that portion of the specifications which sets forth the payload and performance requirements in order to be able to estimate within a very few pounds what the gross weight of the airplane is likely to be as well as what type of wing design is likely to be most suitable. He is able to do this because he has studied similar airplanes; but, even then, his preliminary estimate must be broken down into as many components as possible because an intelligent designer cannot afford to make any estimates without reasonable research to back up those data.

It is necessary to make weight estimates as accurate as possible, for if the airplane should weigh more than estimated, the per-

formance might be seriously affected by the greater weight. If this should happen—the weight be greater than estimated—it might be necessary to cut down on the amount of payload, which certainly is not economical; or to cut down on the fuel load which in turn will cut down the range.

The question may well be asked: "Why cut down on the payload or fuel load if the airplane is overweight; isn't the airplane better if it carries more load than that for which it was originally designed?" No, that is not the case. In any given design, such items as the payload, crew, fuel and oil are fixed in weight; that is, they have been set by the original specification so that any increase in weight is undoubtedly due to the structure which has been either underestimated in weight or improperly designed. In both of these cases the result is that the airplane will cost more to construct, and the increased cost is undesirable. Moreover, the size of the wing is determined by the gross weight and is designed to be strong enough to carry that load. Any increase in weight will also mean, since the wing area cannot be changed, that the airplane will have to land faster, and the faster the landing speed the harder it is for the pilot to handle the airplane. Too high a gross weight will also mean that the airplane will have to travel faster and longer along the ground before it can take off, and unless the runways of the airport are long enough the airplane may not be able to take off at all. Also the heavier the airplane is for the given wing area and the horsepower of the engines, the more difficult it will be to reach a high enough altitude to clear high mountains.

Thus when an airplane is overweight, in order to land slowly enough and take off quickly enough, overloading must be reduced in some way. The only way there is left to do this after the airplane has been built is to cut down the amount of payload or the amount of fuel.

There is another serious factor regarding the overweight of an airplane. When the structure is being designed, each part is designed to withstand a certain load imposed on it under various flying and landing conditions, but if the airplane is overweight some of these parts may fail.

Suppose an airplane has been estimated to weigh 10,000 pounds. The landing gear—taking just one part of the airplane for illustration—is designed to withstand this load without failure when landing. Suppose, however, when the airplane is finally built, that it is

found to weigh 11,000 pounds. Not all of this addition in weight may be due to improper structure design, perhaps, for the designer may have also decided to double the mail or express load after the design was started (much like changing horses in the middle of a stream). The landing gear, originally designed to stand up under 10,000 pounds on landing, must now stand up under 11,000 pounds. Will it do it? It may not.

Suppose the landing gear is designed originally to stand up under 11,000 pounds instead of 10,000 pounds, so that if the airplane does weigh more it will be safe. Unfortunately this is no remedy. For it will be designing a structure which even if not overweight will be unnecessarily overstrong. Overstrength of parts means fundamentally poor and very costly design.

The estimates may be exceeded by a small percentage. A 5 per cent increase in one item may be more than offset by a decrease in weight of another item. In general it is not desirable to exceed the estimate gross weight by more than 2 or 3 per cent. When an entirely new type of airplane is being designed, the final gross weight may be very much larger than the estimate, primarily because insufficient data were available to make it possible to estimate more closely. An airplane designed to carry 250,000 pounds, for example, may weigh well in excess of the allowable 3 per cent, but suppose ten or twelve of this large airplane were built in a series and flown extensively for a few years so that all its characteristics were known; then the next designer to build an airplane of similar size would have some data available and his design should meet the original specification far more closely than the first prototype did.

It is for these reasons that the art of weight estimating is important and interesting. In most airplane factories, there is at least one man known as the Weight Man or Weight Checker whose duty is to weigh every part that goes into the airplane and to keep a careful record. He not only has to keep the record for use in future estimations but also he must see that a particular part intended for a second airplane does not weigh more than that same part which was put in the first airplane. Otherwise each airplane would weigh differently, and therefore behave differently.

The weight estimate goes through a series of trials until the added refinements indicate that it is sufficiently accurate for all subsequent calculations. The weight estimate really does not become "final" until the airplane has been built and each part weighed.

Estimating the Gross Weight

The gross weight, before a more detailed weight breakdown has been made, can first be estimated by means of rule-of-thumb methods.

One method is to determine the probable gross weight on the basis of the power loading since the specifications for the design usually specify the rated engine horsepower for the design. Since

$$\text{Power Loading} = \frac{\text{Gross Weight}}{\text{Total Engine Horsepower}}$$

then assuming a value for the power loading, and knowing the total rated engine horsepower, a reasonable figure for the gross weight may be obtained.

A study of the airplane characteristics, tabulated on page 268, will help in choosing a reasonable value for the power loading for the type of airplane the designer has in mind.

Another method is to estimate as closely as possible the detailed weights for the

1. power plant,
2. crew and payload,
3. combustible load,
4. furnishings and equipment.

These four items constitute from 65 to 70 per cent of the gross weight, or in equation form

$$\text{Gross Weight} = \frac{\text{Sum of Items 1, 2, 3 and 4}}{.7}$$

Therefore a second gross weight can be obtained, and then the difference between the thus calculated gross weight and the above four items will gives the amount allotted for the structural weight.

Estimation of the Structural Weight

It is necessary to figure a likely value for the gross weight of the airplane in order to determine

1. the wing area and consequently the wing weight,
2. the size and weight of the landing gear,
3. the size and weight of the tail surfaces (once the wing area is tentatively determined, the area of tail surfaces are likewise tentatively determined),
4. fuselage length.

Manufacturers' ratings for wheels, tires, shock absorbers and the like are in terms of the static weight on these parts. Therefore if the gross weight of the airplane is known, the static weight per wheel is then one half the gross weight of the airplane. The static weight on the tail wheel varies from 1/10th to 1/12th the gross weight of the airplane. Once these static weights are known, the sizes of the wheels and tires can be found by consulting the catalogues of the manufacturers.

Once the wing area is known, a tentative span length can be determined, and since the fuselage length varies from 60 to 70 per cent of the span, the fuselage dimensions can be readily determined. Just as soon as the fuselage dimensions are at least tentatively established, the weight can be estimated.

Recapitulations

After the structural weights have been listed, the weights are totalled to give another gross weight which may vary considerably from the previous values determined by rule-of-thumb methods. This gross weight will change the wing and tail-surface area and therefore the weights of these items. The weight of the landing gear usually remains the same since the difference in gross weight is usually not so large as to affect the ratings of the landing gear components.

These corrections in the structural weight are continued with every new total for the gross weight caused by these corrections until the changes are sufficiently small to make further corrections unnecessary.

Form for Preliminary Weight Estimate

For convenience the weights of the airplane are grouped as indicated below. This form is shown in its preliminary stage. It may be considerably expanded, and it is usually desirable to do so, especially in presenting the final weight estimate.

TABLE 1

Preliminary Weight Control

Item	*Weight*
Power Plant	
Engines—Bare Weight	
Starters	
Cowlings	
Generators	
Engine Instruments	
Engine Controls	
Exhaust Manifolds	
Propellers and Hubs	
Fire Wall	
Engine Mount	
Fuel Tanks	
Fuel System	
Oil Tanks	
Oil System	
Miscellaneous	
Total	
Combustible Load	
Gasoline	
Oil	
Total	
Crew	
Pilot and co-pilot	
Navigator	
Radio Man	
Mechanic	
Stewardess	
Total	

Payload	
Passengers and Baggage	
Mail	
Express	
Total	
Furnishings and Equipment	
Flooring	
Seats and Cushions	
Surface Controls and Wires	
Batteries	
Instruments	
Lighting	
Heating and Ventilating	
Radio	
Upholstery	
Soundproofing	
First Aid	
Refreshments (Luncheons, Water)	
Miscellaneous	
Total	
Structure	
Wing Group	
Fuselage	
Empennage—Vertical Tail Surfaces	
Horizontal Tail Surfaces	
Landing Gear—Chassis	
Wheels, Tires and Brakes	
Tail Wheel and Tire	
Cowling	
Total	
Total for Gross Weight	

After the installation and structural layouts have been completed, a more detailed form of the weight calculations, as indicated in the table below, should be made out. This table is useful for future reference and comparison.

TABLE 2

Weight Control

Power Plant	
Engines	
Per Cent of Power Plant Weight	
Per Cent of Weight Empty	
Per Cent of Gross Weight	
Propellers and Hubs	
Oil Tanks and Oil System	
Fuel Tanks and Fuel System	
Starters	
Generators	
Engine Instruments	
Engine Controls	
Fuel Pumps	
Scoops	
Switches	
Thermocouples	
Deflectors	
Cowlings	
Fire Walls	
Engine Mounts	
Exhaust Manifolds	
Total Power Plant	
Per Cent of Weight Empty	
Per Cent of Gross Weight	

Combustible Load	
Gasoline	
Oil	
Total Combustible Load	
Per Cent of Disposable Load	
Per Cent of Gross Weight	
Crew	
Pilot and Co-Pilot	
Navigator	
Radio Man	
Mechanic	
Stewardess	
Total Crew	
Per Cent of Disposable Load	
Per Cent of Gross Weight	
Payload	
Passengers and Baggage	
Mail	
Express	
Total Payload	
Per Cent of Disposable Load	
Per Cent of Gross Weight	
Furnishings and Equipment	
Flooring	
Surface Controls and Wires	
Seats and Cushions	
Safety Belts	
Batteries	
Instruments	
Lighting	

Radio Equipment	
Upholstery	
Flares	
Soundproofing	
Heating and Ventilating	
First Aid	
Refreshments	
Miscellaneous	
Total Furnishings and Equipment	
Per Cent of Weight Empty	
Per Cent of Gross Weight	
Structure	
Wing Group	
Wing Panels	
Ailerons	
Flaps	
Landing and Navigation Lights	
Wing Controls	
Struts	
Total Wing Group	
Pounds per Square Foot of Wing Area	
Per Cent of Structure Weight	
Per Cent of Weight Empty	
Per Cent of Gross Weight	
Empennage Group	
Elevators	
Trimmers or Tabs	
Stabilizers	
Rudders	
Trimmers or Tabs	

Fins	
Struts	
Wings	
Control Systems	
Static Balances	
Total Tail Group	
Per Cent of Structure Weight	
Per Cent of Weight Empty	
Per Cent of Gross Weight	
Body Group	
Fuselage, Including Covering	
Doors	
Windows	
Fillets	
Total Body Group	
Per Cent of Structure Weight	
Per Cent of Weight Empty	
Per Cent of Gross Weight	
Landing Gear Group	
Wheels	
Tires	
Brakes	
Shock Absorber Struts	
Additional Struts	
Retracting Mechanism	
Cowlings	
Tail Wheel	
Tail Tire	
Tail Shock Absorber Strut	
Retracting Mechanisms	

Tail Cowling	
Total Landing Gear Group	
Per Cent of Structure Weight	
Per Cent of Weight Empty	
Per Cent of Gross Weight	
Total	

Notes: Weight empty = structure weight plus power plant plus furnishings and equipment
= gross weight minus disposable load
Disposable load = combustible load plus crew plus payload
Gross weight = weight empty plus disposable load

Miscellaneous Weight Calculations

To aid in estimating the weight of the airplane a list of weights has been collected and is presented below.

Structure

Wings

1. Metal wings internally braced, no flaps, 2.2 lb. to 2.6 lb. per sq. ft.
2. Metal wings internally braced, with flaps and retracting mechanism, 2.6 to 3.5 lb. per sq. ft.
3. Fabric covered wings, 1.9 to 2.5 lb. per sq. ft.
4. Hydraulic flap controls including piping and tank, 100-125 lb.

Tail Surfaces

Vary from 1.2 to 2 lb. per sq. ft.
Near the lower figure for externally-braced tail surfaces.
Figure allows for bracing.
Near higher figure for full cantilever tail surfaces.
Static balances increase weights.

Fuselage

1. Welded tubular type, 1.70 to 1.80 lb. per in. of length.
2. Reinforced monocoque, 1.60 to 1.70 lb. per in. of length.
3. Reinforced monocoque weight obtained by formula.
   .792 x (length) x circumference of median section in ft.
4. The length of the fuselage proper is considered to be the distance from the fire wall to the tail post.
   A typical bare fuselage weight (for a reinforced monocoque construction) including firewall, stringers, covering, fittings, built-in supports, doors and door frames for an airplane weighing about 8500 lb. is 650 lb.

Power Plant

Engine mount, including rubber bushings, bolts, etc., 32-40 lb.
N. A. C. A. cowling (for radial engine of about 750 hp.), 45-50 lb.
Nose cowl with shutters, 12-17 lb.
Oil piping, clips, etc., $2\frac{1}{2}$-4 lb. per engine
Fuel piping, clips, etc., 20-30 lb. per engine
Fuel tanks (welded or riveted aluminum or aluminum alloy)
From 10- to 50-gallon capacity, 9-30 lb. (varies linearly).
From 50- to 100-gallon capacity, 30-50 lb. (varies linearly).
From 100- to 200-gallon capacity, 50-80 lb. (varies linearly).
Oil tanks—allow for 10% more volume. Weights as for fuel tanks.
Oil cooler (for 750-850 hp.), 16-20 lb.
Fuel weight, 6 lb. per gal.
Fuel weight may also be calculated by formula:

$$\frac{(\text{Horsepower at Cruising})\ (\text{Range in Miles})\ (\text{lb./hp./hr.})}{\text{Cruising Speed in Miles per Hour}} = \text{Total Weight of Fuel}$$

Oil weight, 7.5 lb. per gal.

Oil weight may be determined by a formula similar to the above or by allowing at least one gallon of oil for every 16 gal. of fuel; or by allowing 10 gal. of oil plus 1 additional gallon for every 20 gal. of fuel. The volume of one gallon is 231 cubic inches.

Engine accessories vary according to size and type and are best estimated as to weight by consulting manufacturers' catalogues.

Fire wall, 7-10 lb.
Fire wall materials—aluminum alloy, 0.101 lb. per cu. in.
Terne plate, .360 lb. per cu. in.
Engine controls—single engine, 10-15 lb.
Cooling system, 0.7 to 1.0 lb. per horsepower
(allow $\frac{2}{3}$ as much for Prestone)
Exhaust manifolds—short stacks, 12-18 lb.
collectors, 30-50 lb.
Prestone 9.3 lb. per gal.

Propeller

Wood—50 hp.—7 ft. diameter, 13-15 lb.
Aluminum alloy—54 hp.—7 ft. diameter, 46 lb.
Wood—100 hp.—$8\frac{1}{2}$ ft. diameter, 19-22 lb.
Aluminum alloy—100 hp.—$8\frac{1}{2}$ ft. diameter, 52 lb.
Aluminum alloy—600 hp.—14 ft. diameter, 2 blades, 208 lb.
Aluminum alloy—800 hp.—$11\frac{1}{2}$ ft. diameter, 320 lb.
(controllable pitch, 3 blades)

Landing Gear

Wheels and tires are rated according to static weights imposed. The static weight per wheel and tire is ½ of the gross weight. The static weight per tail wheel is from 1/10 to 1/12 the gross weight.

For a 9000-lb. Airplane

Landing gear retracting mechanism	65 lb.
Landing gear chassis (two-strut cantilever type), wheels, brakes, axles, tires, etc.	400 lb.
Hydraulic brake operating mechanism	15 lb.
Landing gear fairings	60 lb.
Tail wheel assembly and fairing	35 lb.
Total	575 lb.

For a 19,000-lb. Airplane

Wheels and brakes	188 lb.
Tires and tubes	226 lb.
Axles	37 lb.
Shock absorber struts	210 lb.
Rear brake struts and attending structure	195 lb.
Retracting struts, locks, etc.	45 lb.
Brake operating mechanism	30 lb.
Total	931 lb.

Weights on Percentage Basis

Fuselage, bare weight	10% gross weight
Engine mounts	3-3½% gross weight
Nacelles in wings	1.5-2% gross weight
Landing gear (including retraction mechanism)	5-6½% gross weight
Wing	15% gross weight

Electrical Equipment

Ammeter Weston No. 301 50-0-50	.5 lb.
Ammeter Weston No. 354 30-0-30	.24 lb.
Battery, storage, 8 volts, 9 amps.	9.2 lb.
Battery, storage, 12 volts, 15 amp. hours	25.0 lb.
Container	3.5 lb.
Battery, storage, 12 volts, 35 amp. hours	38.0 lb.
Container	4.8 lb.
Battery, storage, 12 volts, 70 amp. hours	70.0 lb.
Container	7.5 lb.

Box, fuse (3 fuses)	.4 lb
Box, fuse (6 fuses)	.7 lb
Compass lights	.2 lb
Rheostat	.1 lb
Flare, parachute	19.0 lb
Flare bracket	3.0 lb
Generator (25 amp.) engine-driven	20.0 lb
Generator (50 amp.) engine-driven	35.0 lb
Generator (15 amp.) engine-driven	16.0 lb
Generator control box	3.0 lb
Generator (25 amp.) wind-driven with propeller	18.0 lb
Instrument board light	.1 lb
Rheostat	.1 lb
Landing lights (pair)	17.1 lb
Without streamlines (pair)	10.4 lb
Battery	
40 amp. hour	49.5 lb
60 amp. hour	64.5 lb
Switches (pair)	1.3 lb
Light—dome	.25 lb
Magneto—Dixie booster	8.25 lb
Navigation lights (3)	.7 lb
Switch	.1 lb
"Blinker"	.2 lb
Very pistol	2.5 lb

Wire and Cable

Description	*Use*	*Size Number*
High tension	Ignition — magneto to plugs — starting magneto to motor magneto	14 B & S GA
Low tension	Lighting—switches to magneto Ignition—switches to distributor	14 B & S GA
Low tension	Starting—battery to starter	4 B & S GA
High tension	Interior (single wire)	14
High tension	Interior (single wire)	14
Low tension	Miscellaneous connections (single wire)	14
Low tension	Miscellaneous (two wire)	14

Intercommunication and Radio

High tension	Antenna circuits (single wire)	14
Low tension	Antenna circuits (3 wires)	20
Antenna	Antenna and internal bonding (single)	16
Fixture	Compass coil (single wire)	18

Description O. D.	*Number Braids*	*Weight per Foot*
High tension 7 m. m. O. D.	2	.075
Low tension 2 5 m. m. O. D.	2	.040
Low tension .375 in. O. D.	1	.167
High tension 15/32 O. D.	1 layer rubber	.120
	3 layer tape	
	2 layer braid	
High tension 19/32 O. D.	1 layer rubber	.120
	3 layer tape	
	2 layer braid	
Low tension 13/64	1 layer rubber	
	2 layer tape	.025
Low tension 7/32x35/64		.075

Intercommunication and Radio

High tension 5/8	3 layer rubber	.225
Low tension 17/64	1 layer rubber	
	1 varnished cambric	
	1 colored braid	.040
Antenna 1/16	No cover	
	Cotton center	.010
Fixture 1/8	2 layers braid	.0097

Equipment, etc.

Air rafts (1)	24.0 lb
Anchor	200.0 lb
Anchor cable	20.0 lb
Anchor winch	43.0 lb
Ash trays, coat and hat racks, service call systems, etc. per passenger	2.5 lb
Sea anchor and line	10.0 lb
Automatic pilot	110.0 lb
Hand axe	2.0 lb
Baggage—usually allow 30 lb. per passenger	
Bedding, per berth	12-18 lb
Bilge pump	7.0 lb
Boat hook	4.0 lb
Cabin entrance doors	12-15 lb

Chairs, Seats and Cushions	*Weight without upholstering lb.*		*Weight upholstered lb.*
Dural-reclining (single seat)	6.50		12.06
Dural-reclining (double seat)	12.50		24.56
Steel tubing	5.00		8.50
Steel tubing (reclining)	7.00		13.00
Wicker	4.75		
Wicker	8.50		
Wicker (settee)	10.125		
Wicker	8.56		14.43
Wicker	8.75	with cushion	10.75
Rubber air-inflated 17 x 19 air cushion	1.39		
17 x 29 air cushion	2.14		

Chemical toilet	10.5 lb.
Coat rails	5.0 lb.
Cockpit enclosure and fairing	80-100 lb.
Control system including dual rudder, pedals, dual wheel control, stabilizer adjustment mechanism, cables, rods, pulleys and supports	110-120 lb.
Crew and passengers	each 170 lb.
Cupboards	10.0 lb.
Curtains	1.25 lb.
Drinking fountain	36.0 lb.
Engine fire extinguisher	80.0 lb.
Entrance ladder	9.0 lb.
Entrance railing	4.0 lb.
Fire extinguisher	14.0 lb.
First aid kit	3.0 lb.
Fish oil	9.0 lb.
Flash lights	1.75 lb.
Flit and spray gun	2.5 lb.
Floors—metal	per sq. ft. 1-1½ lb.
Floors—wood	per sq. ft. 1 lb.
Fog horn	.5 lb.
Food	100.0 lb.
Food box	23.0 lb.
Hand axe	2.0 lb.
Hat holders	3.0 lb.
Heating and ventilating system, including cabin ducts,	65-100 lb.
allow per passenger	6.5 to 10 lb.
Heaving line	4.0 lb.
Hydraulic operating system for landing gear and wing flaps (for airplane of gross weight of 19,000 lb.)	120 lb.

Weights of Instruments (Pioneer Instrument Co.)

Instrument	*Type No.*	*Wt. Lb.*	*Wt. Kilos*
Air distance recorder	351	0.7	0.317
Air distance transmitter	T4B	1.1	0.500
Air speed indicator 160 or 200 m.p.h., less pitot	354	0.6	0.272
Air speed indicator 300 m.p.h., less pitot	354	0.6	0.272
Air speed indicator rotatable 200 m.p.h., less pitot	735	0.9	0.408
Air speed indicator rotatable 300 m.p.h., less pitot	735	0.9	0.408
Altimeter, one revolution, 10,000 or 20,000 ft.	355C	0.7	0.317
Altimeter, two revolutions, 10,000, 20,000 or 40,000 ft.	743B	0.6	0.272
Altimeter, rotatable, two revolutions, 10,000 or 20,000 ft.	804	0.9	0.408
Altimeter, sensitive, 20 revolutions, 20,000 ft.	399	1.0	0.453
Accelerometer	876	0.9	0.408
Ammeter	782	0.4	0.180
Bank indicator, 10-0-10 degrees	651	0.2	0.090
Bank indicator, 20-0-20 degrees	770	0.4	0.180
Climb indicator—indicator	374B	0.6	0.272
Climb indicator—bottle and tubing		1.2	0.544
Clock, large standard Waltham	543 & 544	0.75	0.340
Clock, large standard Elgin	880	0.75	0.340
Clock, small standard Elgin	757	0.4	0.180
Compass—spherical bowl	718B	1.0	0.453
Compass—spherical bowl	797	1.2	0.544
Compass—spherical bowl	780	2.3	1.004
Compass—spherical bowl	818	3.4	1.542
Compass—cylindrical bowl	383C	2.1	0.955
Compass—cylindrical bowl	518C	3.4	1.542
Compass—Straightflight, Jr.	748	2.25	1.002
Compass—Straightflight	790	5.5	2.500
Compass—Navy Type, Mark VII	726	3.7	1.680
Compass—Navy Type, Mark VIII			
Compass—Earth Induction	301D		
Generator		9.6	4.360
Controller		1.1	0.500
Indicator		0.7	0.317
Flex, shafting per foot		0.06	0.027
Flex, cable per foot (electric)		0.05	0.022
Flex, cable per foot		0.09	0.040

Compass—Bumstead Sun		1.5	0.680
Drift indicator	625B	1.4	0.635
Drift indicator, special lens	753	3.5	1.590
Fuel level gauge	371	0.7	0.317
Cell with 30″ tubing		0.3	0.136
Pump		0.2	0.090
Manifold pressure gauge	791	0.5	0.227
Manifold pressure gauge	808	0.9	0.408
Manifold pressure gauge	814B	0.75	0.340
Manifold pressure gauge	772	1.0	0.453
Octant (carrying case, wt. 3.7 lb.)	342	2.8	1.270
Oxygen regulator	838	3.0	1.360
Pitch indicator	502B	1.4	0.635
Pressure gauge, fuel	751	0.3	0.136
Pressure gauge, oil	505B	0.3	0.136
Suction gauge, (large)	792	0.5	0.227
Suction gauge, (small)	763	0.3	0.136
Tachometer (magnetic)	734	1.0	0.453
Tachometer (centrifugal)	347B	0.9	0.408
Tachometer (electric—2 revolutions)	805		
Indicator—2 revolutions		1.2	0.544
Generator		2.5	1.135
Semi. rotat. indicator		2.0	0.909
Thermometer (6 ft.)	506B	0.8	
Thermometer (12 ft.)	506B	1.1	0.500
Thermometer (20 ft.)	506B	1.3	0.590
Turn indicator	385B	1.2	0.545
Venturi tube	74B-900	0.4	0.180
Venturi tube	V-3F	0.7	0.317
Voltmeter (Weston)		0.4	0.180
Warning units			
Vacuum		0.4	0.180
Fuel pressure	760	0.4	0.180
Watches (navigation)		1.5	0.581
Accessories			
Anti-spill vent	775	0.4	0.180
Flare, parachute Wiley	A-8	19.0	8.640
Flare bracket		3.0	1.362
Fuel pump, hand	173B	2.5	1.170
Light, compass	356-920	0.15	0.066
Light, instrument board	127	0.15	0.066
Oil, Wilkins (5 oz.)		0.3	0.136

Pitot static tube	5F-900	0.3	0.136
Pitot static tube	5H-900	0.5	0.227
Pitot static tube	5J-900	0.3	0.136
Pitot static tube, electric	357E-905	0.7	0.317
Pitot static tube, electric	357D-914	0.9	0.408
Switch, landing light (pair)	S-169	1.3	0.590
Switch, Scintilla magneto (ED, EJ, EF, EE)		0.7	0.317
Switch, Scintilla magneto, twin motor	626	0.8	0.363
Switch, Scintilla magneto, tri-motor	573	0.9	0.408
Tachometer shaft (fabric casing) per ft.		0.03	0.013
End fitting		0.10	0.045
Terminal blocks (2 terminal)	533-2	0.06	0.027
Autosyn Instruments and Accessories			
Autosyn motor	769B	0.625	0.317
Autosyn indicator (small dial)	842 etc.	1.0	0.453
Autosyn indicator (large dial)	887 etc.	1.06	0.500
Autosyn indicator tandem (small dial)		1.7	0.772
Autosyn indicator tandem (large dial)	874 etc.	1.95	0.955
Autosyn fuel pressure transmitter	833	1.65	0.772
Autosyn oil pressure transmitter	832	1.4	0.635
Autosyn manifold pressure transmitter	872	1.7	0.772
Autosyn thermometer transmitter	844	1.75	0.864
Autosyn tachometer transmitter		1.9	0.864
Autosyn fuel flow transmitter	886	4.0	1.820
Autosyn fuel consumption transmitter		1.4	0.635
Autosyn radio tuner (without indicator)	866	2.2	1.000
* * *			
Alternator (small)	848	4.1	1.860
Transformer (110-32) H. C.		2.3	1.004
Mirror (7 x 10 inches)		1.3	
Oil regulators		15.	
Pail (10 quarts)		2.0	
Paper cup holder		2.0	
Paper towel holders		2.0	
Parachutes			
Irving, 24 ft. lap type		18.0	
Irving, 24 ft. slat type		19.5	
Irving, 24 ft. back type		17.5	
Irving,2 8 ft. back type		24.0	
Pilot's seat cushion type		23.5	
Russell, with cushion and back pad		21.0	
Parachute flares		50.0	

Radiator Data

Tubes—Plain Hex. Ends								*Per Sq. Ft. of Frontal Area*	
Size Round Tube	*Overall Length*	*Size Hex. Across Flats*	*Length of Hex.*	*Frontal Area of Tube*	*Sq. In. Cooling Surface of Tube*	*Weight Pounds Each*	*Sq. Ft. Cooling Surface*	*Weight Water*	*Weight Tube*
$\frac{1}{4} \times .006$	4	.313	.187	.08485	3.30	.0064	38.95	6.3	15.9
"	5	.313	.187	.08485	4.11	.0079	48.43	8.0	19.8
"	6	.308	.250	.08214	4.93	.0095	60.07	8.9	23.6
"	7	.308	.250	.08214	5.74	.0111	69.86	10.5	27.6
"	9	.308	.250	.08214	7.35	.0143	89.43	13.8	35.6
"	7	.323	.250	.0903	5.76	.0111	63.76	Air cooler	25.2
"	9	.323	.250	.0903	7.37	.0143	81.58	Air cooler	32.3

Approximately .75 sq. ft. of cooling surface is required per horsepower of motor.

Add approximately 20% to dry core weight for weight of shell and mounting brackets.

Weight of radiator for 100 horsepower engine, 38 to 42 lb.

Weight of oil cooler for 100 horsepower engine, 6 to 12 lb.

Radio complete	92.5 lb.
Radio and mechanic's seat	30.0 lb.
Radio mast antenna	10.0 lb.
Radio table	5.0 lb.
Radio wiring	39.0 lb.
Ring cowlings, each	44.0 lb.
Safety belts, each	1.15 lb.
Sanitary pad holders	1.0 lb.
Ship's bell	1.5 lb.
Smoking stand	4.0 lb.
Soundproofing Materials	0.4 to 0.75 lb. per sq. ft.
(a) Dry zero—2 in. thick	0.19 lb. per sq. ft.
(b) Balsam wool—$\frac{1}{2}$ in. thick	0.18 lb. per sq. ft.
(c) Balsam wool—1 in. thick	0.30 lb. per sq. ft.
Spare parts kit	45.0 lb.
Starter handles	5.0 lb.
Starters	20-35 lb.
Steward's seat	9.0 lb.
Strong box	9.0 lb.
Table rack	7.0 lb.
Tool kit	16.0 lb.
Towel holders	2.0 lb.

Utility ropes	14.0 lb.
Very pistol and cart.	7.0 lb.
Waste can	4.5 lb.
Water	62.4 lb. per cu. ft.—0.25 lb. per glass
Water canteens	10.5 lb.
Water containers	108.0 lb.
Windows and interior frames, rubber moulding, per window	2-2.5 lb.

Material—Miscellaneous

Asbestos (sheet)	.089 lb. per cu. in.
Bakelite	.046 lb. per cu. in.
Artificial leather—upholstering material	.16 lb. per sq. ft.
Canvas—oz. per linear yd. 22 in.—7	.080 lb. per sq. ft.
8	.091 lb. per sq. ft.
15	.170 lb. per sq. ft.
Celluloid—1.16 in. thickness	.47 lb. per sq. ft.
Triplex	2.15 lb. per sq. ft.
Cord, silk, braided—tensile strength, lb.—100	.0013 lb. per sq. ft.
Cord, twisted—tensile strength, lb.—350	.0040 lb. per sq. ft.
Fabric—mercerized cotton airplane fabric, Grade A	.021 lb. per sq. ft.
Felt—$\frac{1}{4}$ in. thickness	.28 lb. per sq. ft.
$\frac{1}{2}$ in. thickness	.57 lb. per sq. ft.
1 in. thickness	1.14 lb. per sq. ft.
Gimp artificial leather—$\frac{1}{2}$ in. width	.0148 lb. per sq. ft.
$\frac{5}{8}$ in. width	.0188 lb. per sq. ft.
Glass—Safetee: super quality 5/32—7/32 in. thick	2.5 (approx.)
plate quality 7/32—9/32 in. thick	4. "
special quality 7/32—9/32 in. thick	4. "
Non-shatterable	.091 lb. per cu. in.
Kapok	.0035 lb. per cu. in.
Leather, genuine	16 lb. per sq. ft., .12 to 176 lb. per sq. ft.
Liquids	
Alcohol	6.6-6.8 lb. per gal.
Benzol	7.0-7.2 lb. per gal.
Ethylene glycol	9.34 lb. per gal.
Gasoline	5.9-6. lb. per gal.
Oil	7.5 lb. per gal.
Water (fresh)	8.3 lb. per gal.
Water (salt)	8.5 lb. per gal.
Landing lights	24.0 lb.

Lavatory and equipment	45.0 lb.
Machete	2.5 lb.
Magazine holders	4.0 lb.
Mail or express—allow at least $12\frac{1}{2}$ cu. ft. for each 200 lb. of mail to be carried.	
Mail bag—with brass padlock, for registered mail	6.5 lb.
Mail bag—iron locked	2.5 lb.

Metals

Aluminum (cast)	.0924 lb. per cu. in.
Aluminum (rolled)	.0978 lb. per cu. in.
Aluminum alloy	.1012 lb. per cu. in.
Brass	.3075 lb. per cu. in.
Cadmium	.3017 lb. per cu. in.
Chromium	.2457 lb. per cu. in.
Copper	.3212 lb. per cu. in.
Iron	.2634 lb. per cu. in.
Iron (cast)	.2605 lb. per cu. in.
Iron (wrought)	.2779 lb. per cu. in.
Lead	.4108 lb. per cu. in.
Magnesium	.0629 lb. per cu. in.
Manganese	.2890 lb. per cu. in.
Mercury	.4909 lb. per cu. in.
Nickel	.3179 lb. per cu. in.
Silver	.3805 lb. per cu. in.
Steel	.2839 lb. per cu. in.
Tin	.2634 lb. per cu. in.
Vanadium	.1987 lb. per cu. in.
Zinc	.2598 lb. per cu. in.

Paints and Finishes

3 coats clear dope 3 coats aluminum dope 2 coats berryloid	5.7 oz. per sq. ft.
1 wash coat of thinner 5 coats clear dope 2 coats pigmented dope 1 coat clear exterior 507 lacquer	5.25 oz. per sq. ft.

1 coat iron oxide primer, .40 oz. per sq. ft.
1 coat anti-fouling paint, .40 oz. per sq. ft.
1 coat navy gray enamel, .32 oz. per sq. ft.
1 coat varnish, .24 oz. per sq. ft.

1 coat oxide primer (coat navy gray enamel), 1.05 oz. per sq. ft.
2 coats spar varnish, .12 oz. per sq. ft.
1 coat aluminum varnish, .35 oz. per sq. ft.
1 coat bituminous paint, .18 oz. per sq. ft.

Plasticele	.049 lb. per cu. in.
Plexiglas	.043 lb. per cu. in.

Pulleys

Moulded Bakelite Micarta Graphite Impreg. Core

Pulleys Outside Diameter	*Cable Size (in.)*	*Weight (lb.)*	*Groove Diameter*	*Groove Depth*	*Weight (lb.)*
$1\frac{1}{4}$ in.	1/16-3/32	.0107	$1\frac{7}{8}$	7/32	.078
$2\frac{1}{2}$ "	1/16-3/32	.055	$2\frac{3}{8}$	7/32	.122
2 "	1/8-5/32-3/16	.048	$1\frac{1}{8}$	3/32	.019
$3\frac{1}{2}$ "	1/8-5/32-3/16	.171			

Pyralin	.049 lb. per cu. in.

Rubber Shock Absorbing Discs	*Compression*	*Lb. Per Cu. In.*
	7,863 lb. per sq. in.	.0353
	23,101 lb. per sq. in.	.0485
	16,035 lb. per sq. in.	.0410
Rubber	.040-.065 lb. per cu. in.	

Elastic Rubber Cord Diameter	*Lb. per Ft.*	
	Shock Absorber Cord	*Exercising Cord*
3/16		.011
1/4		.019
5/16		.029
3/8	.050	.050
1/2	.090	.090
5/8	.142	.142
3/4	.208	

Silk—Japanese 36 in. wide 1.4 to 1.6 oz. per yard

Tape		Width	Weight, lb. per Ft.
	Cotton-herringbone selvage edges	½	.0018
		⅝	.0024
		¾	.0030
	Pinked edge grade A cotton	2½	.0055
	Plain grade A cotton	2	.0050

	Width	Strength Tensile	Weight per Ft.
Webbing, linen	1¾	2800	.094
	1¾	750	.033

Wood

Ash	41 lb. per cu. ft.
Balsa	10 lb. per cu. ft.
Mahogany	34 lb. per cu. ft.
Maple	44 lb. per cu. ft.
Pine, white	26 lb. per cu. ft.
Spruce, eastern	27 lb. per cu. ft.
Spruce, Sitka (western)	27 lb. per cu. ft.
Cork	15 lb. per cu. ft.
Balsa	9.5 lb. per cu. ft.
Basswood	26 lb. per cu. ft.
Birch	44 lb. per cu. ft.

		Approx. Weight, Lb. per Sq. Ft
Mahogany plywood	1/16—3 ply	.22
	3/32—3 ply	.31
	1/8 —3 ply	.39
	3/16—3 ply	.53
	1/4 —5 ply	.77

PROBLEMS

1. The fuel consumption of a 550 horsepower engine is 0.57 pound per brake horsepower per hour. Its oil consumption is 0.050 pound per brake horsepower. Calculate the fuel and oil required for a three-hour flight.

2. Determine the size of a likely fuel and oil tank for the fuel and oil required in problem 1. Allow 10 per cent additional volume for the oil tank.

3. Prepare a preliminary weight estimate for the "Furnishings and Equipment" group for an airplane which is to carry 17 passengers and 3 crew members.

4. Estimate the weight of a retractable landing gear group for an airplane whose gross weight is 15,000 pounds.

5. Calculate the percentage of the total for the group of each item entering in the estimate of the landing group.

6. The wing loading of a monoplane of gross weight of 10,000 pounds is 25. What is the wing area and the weight of the wing?

7. Prepare a preliminary weight estimate of an airplane whose total gross weight is 3600 pounds. The airplane is a private, single engine monoplane seating 3 people.

8. Select an air-cooled and a water-cooled engine of approximately the same horsepower. Determine the weight for the complete power plant for each engine.

III. The Three-View

The three-view of a projected design is composed of the top or plan view, the front view or front elevation, and the side view or side elevation. It corresponds to the photographs that might be taken of the top, front and side view of the completed airplane.

After the designer has his set of specifications, he makes a few sketches of what he believes his final design should look like. It helps him to visualize arrangements more readily and it forms the basis of his detailed weight estimate and subsequent balance calculations.

An experienced designer will first make a hasty "thumb-nail" three-view sketch. It is sufficiently detailed to convey the basic ideas of the design and no further three-view is then made until the design has been almost completely decided upon as to dimensions, correlation of wing, engine, landing gear and tail surfaces.

However, even the experienced designer usually finds it desirable to make several three-views—each one more accurate than its predecessor.

The "thumb-nail" sketch is excellent to determine the type, to initiate the design and to record particular features of exceptional nature that the designer has in mind. However, the first design on which such work as the weight estimate and balance diagram can be based is the preliminary three-view.

Unless the airplane is radical in design—departs wholly from conventional design—the data obtained from existing airplanes are the best guides in proportioning the new project, and the new or uninitiated designer should avoid "radical" or new designs until he has first worked through a conventional design. The fundamental principles of airplane design always hold, and there is no better way to understand them than by working through conventional designs. "Radical" designs are usually "radical" because they disregard fundamental principles with the result that the designer will eventually be disappointed.

The outline below shows how standard data on existing airplanes may be made of use in laying down the preliminary three-view.

In making a preliminary three-view, some knowledge of the installation requirements of the power plant, cockpit and cabin is necessary; the chapters on these subjects should be carefully studied before going too far with the preliminary sketch.

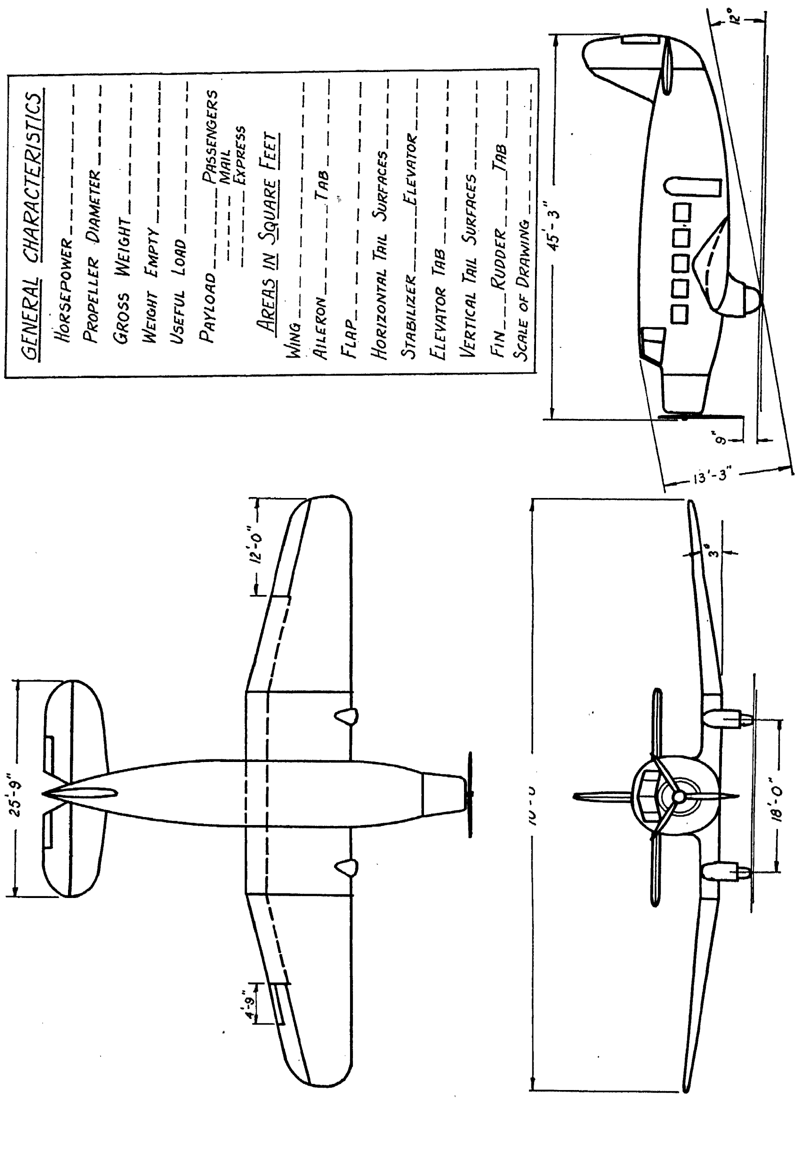


FIGURE 1. PRELIMINARY THREE-VIEW

Steps in Assembling Data

The information necessary for the three-view can be assembled easily after the specifications for the airplane are given. These specifications, when compared with similar existing airplanes, indicate the probable size of the airplane, as well as the relative ratio of tail surface areas, lengths, wheel tread and the like to the wing area and wing span. The wing area and its dimensions may be considered the starting point of this preliminary three-view.

The steps, in sequence, are given below.

1. *Estimation of Wing Area*

The wing area may be determined if the wing loading is assumed. For example if a wing loading of 20 pounds per square foot is assumed then from the relationship,

$$\text{Wing loading} = \frac{\text{Gross Weight}}{\text{Wing Area}}$$

the probable wing area may be determined.

If this method seems unsuitable, another method is to use the known characteristics of the airfoil to be used and determine the wing area by means of the formula:

$$S = \text{Wing Area} = \frac{\text{Gross Weight}}{\frac{1}{2}\ \rho\ C_{L_{max}}\ V^2}$$

where V is the stalling speed, assumed equal to the specified landing speed or $V = 65 \times 1.467$ in feet per second.

ρ = .002378 at sea level where landing usually takes place.

$C_{L_{max}}$ = the maximum lift coefficient for the airfoil to be used.

If different airfoils are used for the root and tip chord, the average value for the maximum lift coefficient for the root and tip airfoils may be used, unless actual wind tunnel tests on the particular wing are available.

In using data for airfoil characteristics, it is suggested that those obtained at Reynold's numbers of about 3,000,000 or over should be used.

If lift increase devices are used, such as partial span flaps, the maximum lift coefficient of the airfoil used may be increased 40 per cent as a preliminary estimate.

2. *Aspect Ratio*

The aspect ratio of a full cantilever monoplane, for example, is between 6 or 7. Assuming an aspect ratio, the span of the design may be determined from:

$$\text{Aspect Ratio} = R = \frac{(\text{Span})^2}{\text{Wing Area}}$$

3. *Planform*

With the wing area decided upon, the aspect ratio determined, a suitable planform may then be selected. If the wing is rectangular, the matter is easy. If the planform is tapered, the tip chord may vary from $\frac{1}{2}$ to $\frac{2}{3}$ of the root chord.

When the approximate dimensions of the chord have been determined, then other items relating to the planform may be decided upon, such as whether (a) the leading edge should be perpendicular to the longitudinal plane of symmetry of the airplane, (b) the trailing edge should be perpendicular to the longitudinal plane of symmetry, (c) the front spars (which are usually located at constant percentages from the leading edge) should be perpendicular to the longitudinal plane of symmetry, or (d) the maximum forward center of pressure line should be perpendicular to the longitudinal plane of symmetry.

4. *Tail Surfaces*

The areas of the vertical and horizontal tail surfaces can be determined from the ratios of these areas to the wing area compiled for existing airplanes.

5. *Landing Gear*

The wheel sizes may have been determined already from a preliminary weight estimate or may be selected now from data furnished by the manufacturers of such equipment. The static weight per wheel is one half of the gross weight. The static weight on the tail wheel is from $\frac{1}{12}$ to $\frac{1}{10}$ the gross weight. It may be more accurately determined after the design has progressed farther.

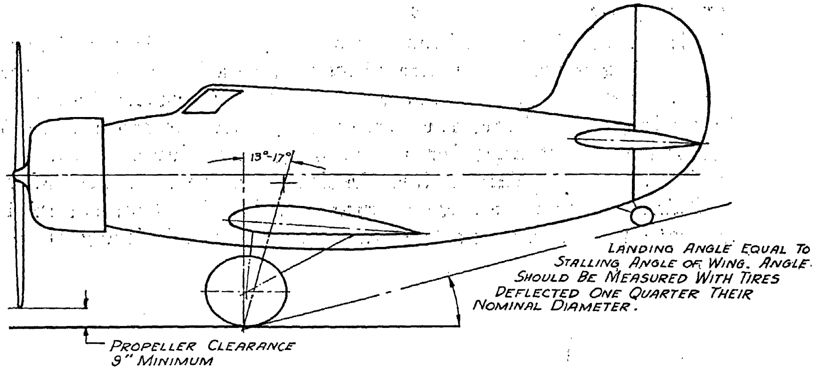


FIGURE 2. CLEARANCES AND ANGLES FOR PRELIMINARY WORK ON THE SIDE VIEW

6. *Clearances*

There are several rules regarding clearances of the propeller with the ground, and of deflected elevators in the three-point landing position with the ground.

The propeller, for instance, should have a minimum ground clearance of 9 inches when the airplane is in the horizontal or level landing position with the shock absorbers and tires deflected as they would be under the normal gross weight of the airplane. For all normal considerations, the shock absorber may be assumed to be deflected about $\frac{2}{3}$ of its normal travel and the tires about $\frac{1}{4}$ of their normal travel. Proper allowance must be made for the configuration of the landing gear when the members are so disposed that the deflection of the shock absorber may cause a greater deflection of the landing gear.

Propellers on seaplanes should clear the water by at least 18 inches when the seaplane is at rest.

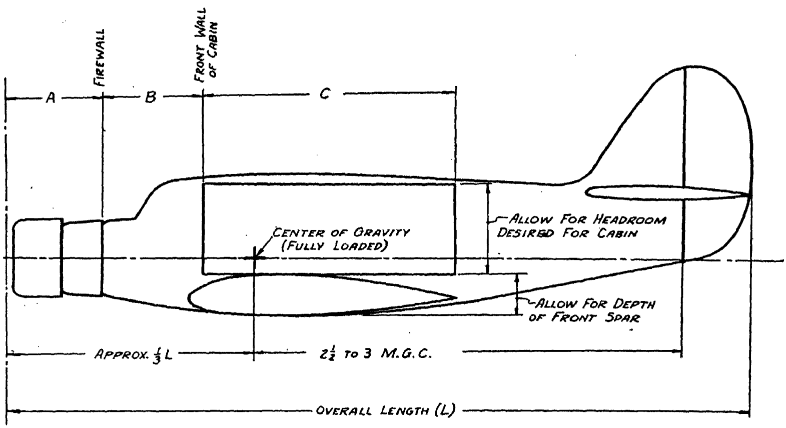
A clearance of at least one inch between the tips of propellers or any part of the structure also should be allowed.

It is generally not wise to allow more than 9 inches clearance for the propeller since this will tend to increase the length of the landing gear members and thereby increase the weight of the landing gear structure.

The elevators should clear the ground by at least 2 or 3 inches when the airplane is in the three-point landing position or at rest.

7. Center of Gravity Location

For the location of the various parts such as the wing, tail surfaces and landing gear, which are placed in relation to the center of gravity, it is desirable to assume the center of gravity location. For a low wing monoplane, the center of gravity is from 2 to 6 inches below the thrust line; for a high wing monoplane, it is about 2 to 4 inches above the thrust line. In lieu of more accurate information obtained from a balance diagram, such a location is sufficiently accurate to assume for the purposes the preliminary three-view is to serve.



A – Overall Length Required for Propeller Hub, Engine, and Engine Accessories with Sufficient Clearance Allowed for Removal of Rearmost Accessory.
B – Distance Allowed for Pilots' Cabin.
C – Length of Cabin Approximately 30(N+1), Where N Represents the Number of Rows of Seats and 30 inches is the Distance Allowed Between Rows.

FIGURE 3. METHOD FOR DETERMINING APPROXIMATE LENGTH OF AIRPLANE FOR THE PRELIMINARY THREE-VIEW

Final Three-View

The final three-view is based upon more accurate information than the preliminary three-view since the weight estimate has been more accurately determined and the final balance diagram has been completed. Both the balance diagram and the final three-view depend in some degree on the structural layout of the wing, landing

gear and fuselage, but once these difficulties in structural arrangements and the like have been ironed out, it is possible to go ahead with the final three-view.

It may be found that the balance diagram has made changes in the following:

1. Position of the wing.
2. Location of the landing gear.
3. Location of the tail surfaces.
4. Location of the center of gravity.

The structural layouts may have caused changes in:

1. Planform of the wing, perhaps because the spar intersected the fuselage where a bulkhead was found to be undesirable.
2. Arrangement of landing gear members due to changes in spar locations.
3. Placement of tail surfaces with respect to planes of symmetry.
4. Vertical position of engine due to installation and vision requirements.

Only after all points have been considered should the final three-view be made. Usually it is desirable to wait until the control surfaces and landing gear, as well as cabin installations, have been made in order to incorporate the latest correction.

PROBLEMS

1. The span for a wing having 420 square feet is 50 feet. With this information and data obtained from the condensed airplane data sheets, calculate the approximate dimensions for:

a. length
b. overall height
c. chord
d. wheel tread
e. vertical tail surface area
f. horizontal tail surface area
g. aileron area

2. Draw a preliminary three-view of the airplane for which the above data was calculated.

8. *Recapitulation*

To aid in drawing up the three-view, a table as shown below may be made, listing all the important data.

Wing		
	Area	sq. ft.
	Aspect Ratio	
	Span	ft.
	Root Chord	ft.
	Tip Chord	ft.
	Root Airfoil	
	Tip Airfoil	
	Mean Geometric Chord	ft.
	Location of 25% of Mean Geometric Chord Projected on Root Chord	ft.
	Aileron Area	sq. ft.
	Aileron Span	ft.
	Aileron Chord	ft.
	Flap Area	sq. ft.
	Flap Span	ft.
	Flap Chord	ft.
	Aileron Area / Wing Area	%
	Flap Area / Wing Area	%

General Dimensions (in feet or in inches)

Overall Span

Overall Height/Overall Span

Wheel Tread/Overall Span

Overall Length/Overall Span

Overall Height

Wheel Tread

Overall Length

Distance between Center of Gravity and Tail Post

All obtained from averages of these same ratios compiled for existing airplanes of similar type.

Tail Surfaces

Vertical Tail Surfaces/Wing Area		*All obtained from averages of these same ratios compiled for existing airplanes of similar type.*
Horizontal Tail Surfaces/Wing Area		
$\frac{\text{Fin Area}}{\text{Vertical Tail Surfaces}} =$	%	
$\frac{\text{Rudder Area}}{\text{Vertical Tail Surfaces}} =$	%	
$\frac{\text{Elevator Area}}{\text{Horizontal Tail Surfaces}} =$	%	
$\frac{\text{Stabilizer Area}}{\text{Horizontal Tail Surfaces}} =$	%	
Horizontal Tail Surface Area		
Elevator Area		
Stabilizer Area		
Aspect Ratio of Horizontal Tail Surfaces		
Span of Horizontal Tail Surfaces		
Chord of Horizontal Tail Surfaces		
Vertical Tail Surface Area		
Fin Area		
Rudder Area		
Aspect Ratio of Vertical Tail Surfaces		
Span of Vertical Tail Surfaces		ft.
Chord of Vertical Tail Surfaces		ft.

Angles

Landing Angle of Airplane

Angle in Side Elevation between vertical through axle and line connecting center of gravity and line joining the center of gravity and the point of contact with the ground at the outer wheel.

Clearances

Propeller Clearance	in.
Deflected Elevator Clearance	in.

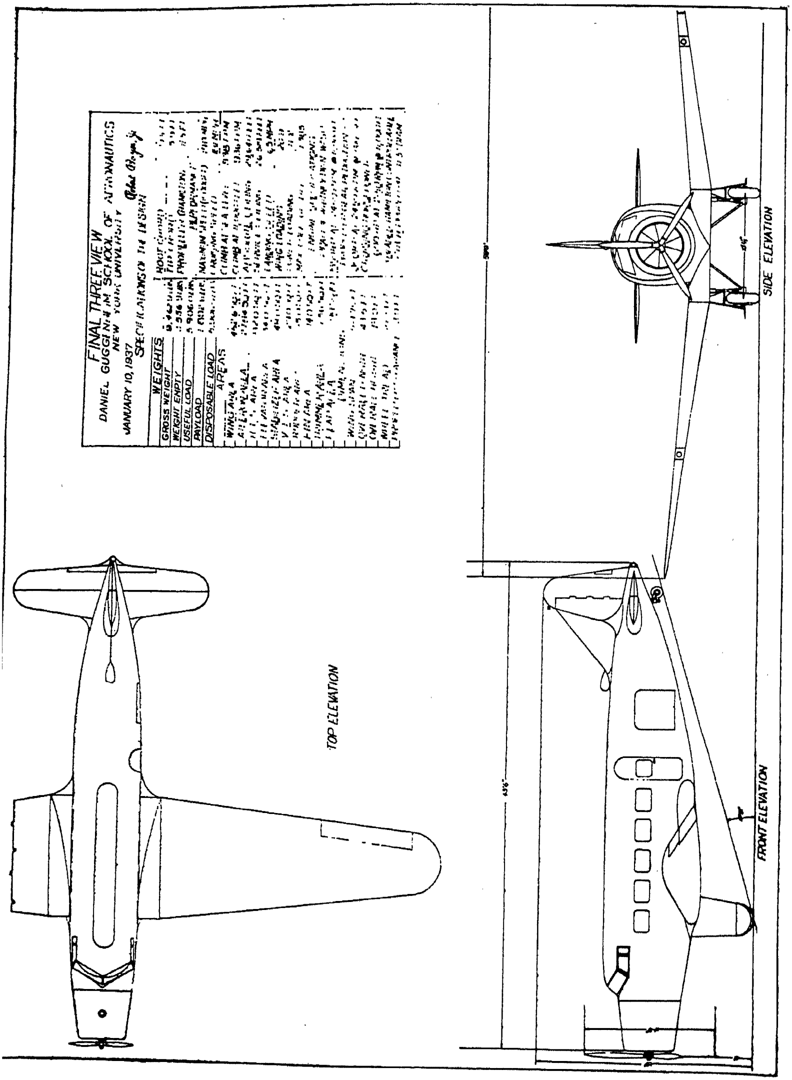

FINAL THREE VIEW
DANIEL GUGGENHEIM SCHOOL OF AERONAUTICS
NEW YORK UNIVERSITY
JANUARY 10, 1937
WEIGHTS
GROSS WEIGHT
WEIGHT EMPTY
USEFUL LOAD
PAYLOAD
DISPOSABLE LOAD
AREAS
TOP ELEVATION
FRONT ELEVATION
SIDE ELEVATION

FIGURE 4

IV. The Balance Diagram

Those who have flown small airplane models know how important the balance of the model is. If there is too much weight in the forward end or "nose" of the model it has a tendency to dive into the ground; if there is too much weight in the rear portion or "tail," it has a tendency to "slide down" backwards with the tail first. For nearly similar reasons, balance is equally important in the full-size airplane. As a matter of fact, the balance is even more important since the flying characteristics are greatly affected by the balance, and in conventional airplanes of today it is customary to place the center of gravity (where the weight of the complete airplane may be assumed to be concentrated) at about the quarter point of the wing chord.

In an airplane the center of gravity, or "balance point," may change while in flight. For example, the fuel and oil are being used up at a constant rate therefore some of the weight disappears, so that the center of gravity is changed. Another case may occur when the balance is correct with the passenger cabin filled—but what happens when there are no passengers, or only a few? Suppose again, that there are only a few passengers in a large cabin and when the flight starts, they sit in the forward portion of the cabin, but later decide to sit in the rear?

These possible changes of what may be called the "disposable load" (although rightly the crew is included ordinarily in the term) have to be considered when the design of an airplane is contemplated.

The balance diagram is such a study. It starts as a preliminary installation diagram, and therefore the fuselage becomes the first object in airplane design, since it normally houses all the items that may change in character, number or location.

In a single engine passenger airplane, for example, with the engine in the nose of the fuselage, the procedure in obtaining a likely balance diagram is to draw the engine and engine accessories accurately to scale, then the firewall, then the pilot's cockpit, then the passenger cabin—in the order in which they normally occur. They are arranged according to the designers' wishes and drawn accurately to scale.

This arrangement or installation serves several purposes. It is similar to the designing of a house by an architect who knows that he has to place the kitchen, bedrooms, bathrooms and the like to provide the maximum of comfort and convenience. At the same time these arrangements help to determine certain dimensions of the house. So with the airplane. The placement of the engine compartment, the pilot's cockpit and the passenger cabin will affect the length of the fuselage and its size and weight.

After the interior arrangements have been placed more or less as they are wanted, a preliminary investigation is made to find out how much the center of gravity of the fuselage and its contents vary when a certain number of passengers are removed. It may be found that because of first dictates for passenger comfort, too much space has been allotted between seats so that when the rear seats of the cabin are empty the center of gravity is too far forward; or the reverse is true when the front passenger seats are empty. Thus the seats should be a little closer so that having a few seats empty will not make so great a difference.

It takes a little practice with each type of design to obtain the right combination. The general rule is that the center of gravity of the completed airplane should not change more than 8 per cent of the mean geometric chord of the wing between the fully loaded condition and a condition corresponding to the most rearward or most forward position of the center of gravity (not necessarily the fully empty condition). This means that for an airplane weighing 10,000 pounds, fully loaded, which has a mean geometric chord of 100 inches, the movement for a condition less than fully loaded should not be more than 8 inches. When the fuselage and its contents are considered alone (as they would be in preliminary calculations) this movement for the same airplane may be greater, say 12%, or 12 inches in this case.

Very often, the designer, over-generous in allowing for passenger comfort, finds that giving too much fore and aft room makes for poor balance conditions for the various possible flight loading conditions. Likewise allowing too much height for the passengers may increase the cross-sectional area of the fuselage so much that performance eventually will be impaired.

The top view is usually left to the last, unless the passenger accommodations are not in rows.

It is usually wise not to give too much space for the engine compartment, or the pilot's cockpit, or the passenger cabin. Especially in the first solution is it desirable to keep to minimum dimensions to avoid grief. Unfortunately, generosity is denied the designer. He must learn to take advantage of many small factors instead of a few large ones. To the uninitiated, the balance diagram may seem simple, but it is really the crux of the entire design. If the rules laid down subsequently in this chapter are obeyed, there is comparatively easy sailing ahead.

After the preliminary three-view and preliminary weight estimate one proceeds to the preliminary balance diagram. This includes part of the installation diagram or inboard profile as well, because arrangements have to be substantially correct to assure a balance diagram of reasonable accuracy.

The Fuselage and Its Contents

Starting with the fuselage and its contents, each item from the nose of the fuselage rearward is placed carefully to scale.

When the fuselage and its contents are located, the center of gravity of the fuselage and its contents is determined. In the calculations to decide upon the center of gravity the horizontal and vertical tail surfaces should be included as factors.

To determine the center of gravity, two datum lines are chosen —one through the center of the propeller in case of a single engine airplane or tangent to the foremost point on the airplane for the horizontal arms of the individual items—another, usually the ground line, for the vertical arms. It is desirable so to choose these datum lines that the arms are all of the same signs and thus avoid possible errors in addition.

Figure 5 shows a partially completed balance diagram indicating the center of gravity location of each item. Sometimes a series of such diagrams is made until the final center of gravity of the whole group is located where the designer wants it.

The calculations should be set up in the form of a table, listing all items included in the fuselage and its contents. Table 3 has been set up as a guide for the balance calculations to be made.

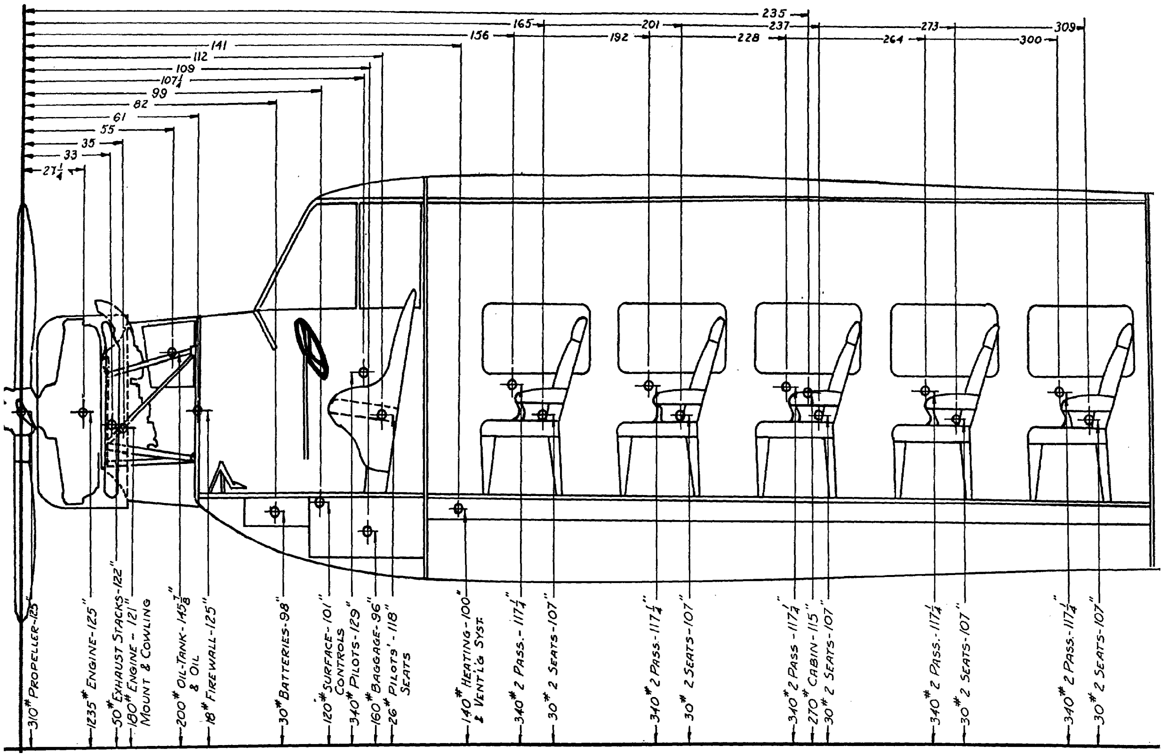


FIGURE 5. A PARTIALLY COMPLETED BALANCE DIAGRAM SHOWING PROCEDURE

The diagram becomes more refined and detailed as the design progresses

TABLE 3

Item	*Weight in Lb.*	*Distance from Vertical Datum*	*Horizontal Moment*	*Distance from Horizontal Datum*	*Vertical Moment*
1. Propeller					
2. Engine					
3. Starter					
4. Hotspot					
5. Airscoop					
6. Engine Cowl					
7. Engine Mount					
8. Oil Tank					
9. Oil					
10. Instruments					
11. Pilot					
12. Co-pilot					
13. Pilot's Seats					
14. Controls					
15. Baggage (fore)					
16. Two Seats					
17. " "					
18. " "					
19. " "					
20. " "					
21. " Passengers					
22. " "					
23. " "					
24. " "					
25. " "					
26. Stewardess					
27. Cabin Furnishings					
28. Radio Equipment					
29. Lighting					
30. Baggage (Rear)					
31. Fuselage Structure					
32. Vertical Tail Surfaces					
33. Horizontal Tail Surfaces					
34. Tail Wheel and Gear					
35. Wing					
36. Fuel Tanks					
37. Fuel					
38. Landing Gear					
Totals					
Column	1	2	3	4	5

Column 1 = sum of all weight items

Column 3 = sum of all horizontal moments

$$\text{Column } 2 = \overline{H} = \frac{\Sigma M_H}{\Sigma W} = \frac{\text{totals in column 3}}{\text{totals in column 1}}$$

Column 5 = sum of all vertical moments

$$\text{Column } 4 = \overline{V} = \frac{\Sigma M_V}{\Sigma W} = \frac{\text{totals in column 5}}{\text{totals in column 1}}$$

It should be noted that it is convenient to lump together small items located at about the same spot and determine the center of gravity of the group by inspection, otherwise determination of the center of gravity of the airplane becomes a series of tedious calculations. The error that may be caused by such a method is certainly not large, as a few simple calculations will demonstrate to the most skeptical.

Locating Wing and Landing Gear

When the center of gravity of the fuselage and its contents has been determined, it is necessary to locate the wing or wings and the landing gear. This work may be done in two steps, that is locate the wing first and then the landing gear, but a few relocations are usually necessary in such a procedure. Usually it is easier to treat the wing and the landing gear as a separate unit, and determine the center of gravity of this unit before attaching the fuselage. Therefore, the landing gear should be located in relation to the wing first, as shown in Figure 6. The final location of the center of gravity of the airplane, fully loaded, is directly above (in a low wing monoplane) or directly below (in a high wing monoplane) the quarter point of the mean geometric chord, projected onto the root chord for convenience.

This clue helps to locate the landing gear in relation to the wing. The distance of the center of gravity of the fuselage and its contents (see Table 3) above the assumed datum line is known as well as the approximate location of the centers of gravity of the wing, its contents, and the landing gear below so that the vertical position can be approximately determined.

Approximate vertical position of C. G.

$$\overline{V}z = \frac{Wy + W_1\ y + W_2\ y_2 + W_3\ y_3}{W + W_1 + W_2 + W_3}$$

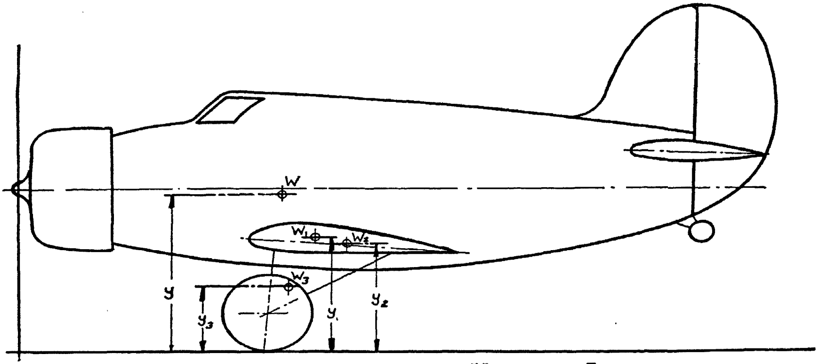


FIGURE 6. DETERMINING THE VERTICAL LOCATION OF THE CENTER OF GRAVITY

where Wy is vertical moment of fuselage and its contents obtained from Table 3 (includes items 1-34 inclusive) or total of column 5.

W_1 = contents of wing
y_1 = approximate distance of wing center of gravity of contents above datum line
W_2 = weight of wing
y_2 = approximate distance of center of gravity of wing above datum line.
W_3 = weight of landing gear
y_3 = approximate distance of landing gear center of gravity above datum line.

These calculations help to locate the landing gear with reference to the wing. It is now time to determine the fore and aft position of the center of gravity of this group.

Assume the leading edge of the root chord as the point of reference.

A subsidiary balance table with reference to Figure 7 may be made as shown in Table 4 below.

TABLE 4

Item	*Weight*	*Horizontal Arm*	*Horizontal Moment*
Wing Fuel Tank Fuel Landing Gear			
Column	1	2	3

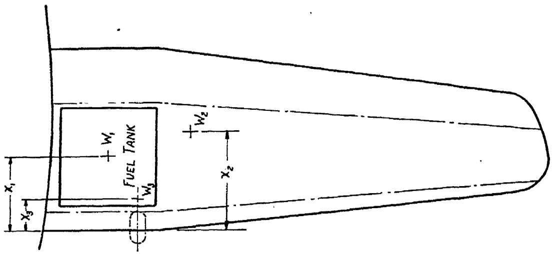


FIGURE 7. DETERMINING THE HORIZONTAL POSITION OF THE CENTER OF GRAVITY OF THE WING, ITS CONTENTS, AND THE LANDING GEAR

Column 1 = sum of all weight items
Column 3 = sum of all horizontal moments

$$\overline{H}_2 = \frac{\text{Column 3}}{\text{Column 1}} = \frac{W_1 x_1 + W_2 x_2 + W_3 x_3}{W_1 + W_2 + W_3}$$

$$c = \tfrac{1}{4} C - \overline{H}_2$$

These calculations finished, the group may be located on the fuselage.

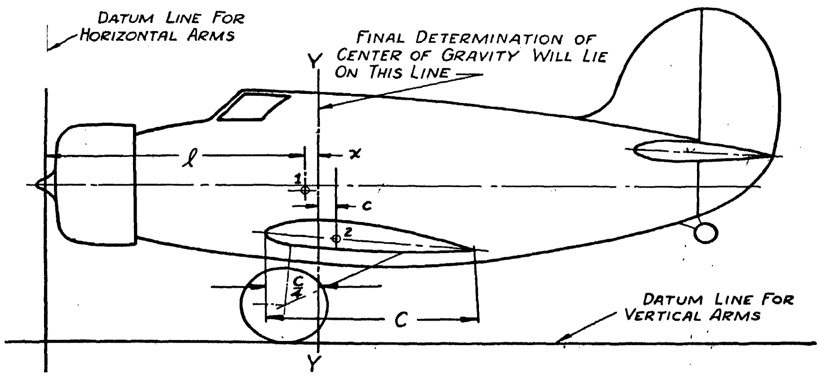


FIGURE 8. DETERMINING THE FINAL LOCATION OF THE CENTER OF GRAVITY OF THE WHOLE AIRPLANE

In Figure 8, *1* indicates the center of gravity of the fuselage and its contents. Its horizontal moment is known and may be found in Table 3.

2 indicates the center of gravity of the wing and landing gear group which has just been determined. Its distance c behind the quarter point on the mean geometric chord C is also known.

The final position of the center of gravity of the complete plane must be in the plane Y-Y through the quarter point just mentioned.

Let W_1 = weight of fuselage and its contents
W_2 = weight of wing and landing gear group

Since horizontal moment of the complete airplane equals the sum of the horizontal moment of the component parts, it follows that:

$$(W_1 + W_2)\ (l + x) = W_1 l + [W_2\ (l + x + c)]$$

from which $W_1\ x = W_2\ c$ is obtained

so that $x = \dfrac{W_2\,c}{W_1}$ giving the necessary information to locate the wing on the fuselage.

Before proceeding with the final form of the balance diagram, check for ground clearance the following items:

1. Depressed elevator
2. Depressed flap
3. Propeller

Also check the distance between the tail post and the center of gravity in terms of the mean geometric chord. All weight items appearing in the original weight estimate should appear in the balance table.

When the balance diagram has been made in its final form, the balance table given in Table 3 should then be corrected accordingly.

Supplementary Calculations

Before the balance is finally accepted, supplementary calculations are necessary to determine the positions of the center of gravity for various possible loading conditions—fully loaded and partially or all empty. The extreme movement of the center of gravity should be limited. A total horizontal movement of about 8 per cent of the mean geometric chord is permissible and, if possible, half of this movement should be ahead and half behind the center of gravity for the fully loaded position.

If the movement is extreme, rearrangement of *variable,* not *fixed,* items such as passengers, baggage, fuel (but excluding the pilot in these calculations) may be made to reduce the movement. Should the center of gravity movement still be beyond the desirable limits, ballast should be provided.

Ballast

Regulations at the present time stipulate:

1. Ballast may be used to enable aircraft to comply with the flight test requirements as to longitudinal stability, balance and landing with the following provisions:
   a. Ballast shall not be used for this purpose in aircraft having a gross weight of less than 5000 pounds, nor in aircraft providing a total seating capacity of less than 7 persons.
   b. It shall be demonstrated by the manufacturer that the specific aircraft can be landed safely without ballast, passengers or cargo.
   c. The place or places for carrying ballast shall be properly designed and installed and plainly marked.
   d. The loading schedule which will accompany each license issued for an aircraft requiring special loading of this type shall be conspicuously posted either in the pilot's compartment or adjacent to the ballast compartments and strict compliance therewith will be required of the aircraft operator.

Centers of Gravity of Individual Items

The centers of gravity of the individual items may be determined by inspection, or rough calculations, or from known locations.

The following list will give some indication of the procedure to be followed.

Propeller	at centerline
Engine	given by manufacturer
Accessories	estimate
Person seated	at bottom vest button
Seat	seat level, forward of back
Furnishings, soundproofing	at geometric center of cabin
Flooring	at geometric centroid
Doors	at geometric centroid
Windows	at geometric centroid
Tail surfaces	center of gravity of each chord section is about 30-35% of chord
Wing	at 40-42% of each chord section; make allowance for flaps
Fuel and oil tanks	at geometric centroid
Landing gear	by inspection

If the wing is assumed to weigh the same per square foot from root to tip, then if the center of gravity of each chord is located 40 per cent behind the leading edge, the center of gravity of the entire wing would be at 40 per cent of the mean geometric chord. Its vertical position may be assumed halfway between the top and bottom camber.

If tail surfaces have irregular planforms, sufficient accuracy will be obtained by resolving the areas in equivalent rectangles and triangles.

Fuel tanks, filled with fuel, of irregular cross-section from one end to the other may be resolved into a series of volumes whose centers of gravity can be determined readily, and it is an easy matter to find the center of gravity of the complete unit. The center of gravity of the empty tank, unless more accurate data is available, may be assumed to coincide with the center of gravity of the tank when filled.

PROBLEMS

1. Determine the center of gravity position of a cabin, due to spacing of passengers alone, with 10 passengers seated side by side, when spaced:

   a. 32 inches apart
   b. 36 inches apart
   c. 40 inches apart

2. What is the importance of determining the center of gravity of an airplane?

3. Determine the effect upon the location of the center of gravity of an airplane whose gross weight is 10,000 pounds and whose center of gravity is 100 inches from the reference line when a total weight of 230 pounds with its center of gravity 156 inches from the reference line has been removed.

V. The Pilot's Cockpit

The cockpit may be considered the most vital part of the airplane, for in it all functions of the airplane and its power plant are coordinated and directed by the pilot and his aides.

All control systems terminate in the cockpit; all operational and navigational instruments are located here; all decisions regarding the flight of the airplane, with the very few exceptions when the airplane is about to take off and land, are determined here. As the airplane grows larger, the cockpit assumes even greater importance. At present it is even more important than the captain's bridge of an ocean liner.

Since the pilot and his aides, such as the co-pilot or the navigational officer or some other member of the crew, spend their entire time on a flight in the cockpit, it is absolutely necessary that every means for comfort, for ease of operation of controls, for coordination of instruments, for vision, and a host of other odds and ends which contribute to the efficiency and well-being of the crew be carefully planned and arranged. Too often an airplane, otherwise satisfactory, cannot overcome the enormous sales resistance caused by a poorly designed or arranged cockpit. The pilot may find that he has not been given enough legroom or enough headroom so that even on moderately short flights, he is easily tired due to a cramped position. Or he may find that his vision forward, sideward, upward and downward is very poor so that when he takes off from an airport, or wants to land, he must guess instead of being able to see how or where he is taking off or landing.

Another difficulty may be that the instruments are not arranged properly: that he cannot quickly see the instruments which he needs to guide him in the proper operation of the engines and in the proper execution of his flight maneuvers.

In some cases minor faults may be corrected to suit the flying personnel, but if, for example, there is insufficient headroom in the cockpit, it may not be possible to make changes unless the airplane is completely re-designed.

Naturally, the thought arises: why not standardize the cockpit in its dimensions, appointments and general arrangements? In other words, design a satisfactory cockpit once and for all. Unfortunately it is not possible to standardize cockpit design wholly until all the

parts and equipment located in the cockpit have been standardized.

Each day sees new instruments which record or indicate additional data not heretofore measured; or new equipment which will take care of some additional functions, but may not wholly replace other equipment of almost but not quite similar functions. The additions to the already available list of instruments and equipment are necessary in order to lighten the burdens of the operating personnel, but their placement may well upset the planned installation of the standardized cockpit. Each new cockpit, therefore, offers its own problems.

However, there is standardization to some degree and this helps enormously in reducing the many problems attendant in cockpit arrangement.

Since the pilot is the most important item in the design of his headquarters, the cockpit is planned around him. To obtain a better conception of his requirements, a celluloid figure, with joints, is made to scale to which the cockpit installation is to be drawn. This jointed figure can then be placed in various postures to see whether the pilot's seat is far enough up from the floor, that the legs are not outstretched too far for comfort, and that the control stick or wheel is not too far ahead so that it is awkward to operate. Of course slight adjustments are obtainable in raising or lowering the seat or in moving it forward or rearward.

If there is insufficient headroom for the pilot to stand up in the cockpit, then there should be at least enough room above his head so that he can lean forward or back, or raise himself up slightly from the seat without fear of bumping his head.

Once the pilot has been comfortably seated—and it is desirable that his legs make an angle of not greater than 45 degrees with the floor to assure comfort in flight—it is necessary to locate the seat so that he may have sufficient vision in all possible directions, with special attention to forward, upward, downward and sideward vision. This done, the windshield may be located. It should not be too far forward since the ceiling will come too far ahead of the pilot and interfere with his line of vision. Likewise the windshield should not be curved in more than two directions because distortion of the scene may result.

Next the instruments and equipment must be considered. Instrumentation is discussed in detail in a subsequent chapter, but reference to a few required features may well be made here. All

switches and controls which the pilot must operate should be located as close to him as possible so that it will be unnecessary for him to stretch. Stretching reduces the pilot's ability to operate the airplane properly at any time and especially in an emergency. On the other hand, those instruments which do not have to be operated by hand may be farther away, but they must be grouped according to functions. The more important functional group are located in front of him, the others slightly out of his prevailing line of direct vision.

When the arrangements for distances and clearances have been settled above and below the pilot, as well as behind and ahead of him some attention should be given to "elbow" room. Here again, he should be given as much room as possible. If there is a co-pilot the width should be doubled and an appreciable allowance made for the aisle between the two. These allowances often determine the width of the fuselage unless the airplane is very large. The result is that designers tend to skimp on width allowances because the greater the width, the greater the eventual effect upon the speed performance of the airplane. However, a difference of 2 or 3 inches will be enormous as far as the comfort of the pilot is concerned, and will hardly affect the speed characteristics of the airplane.

In new designs it is usually the custom to lay out the pilot's cockpit on paper, as discussed above, with the aid of additional information given hereafter, and then to construct a "mock-up" which is a full scale representation of the actual installation in wood, metal and cardboard nailed, bolted or screwed together. In the "mock-up," a man may sit in the "cockpit" and test the arrangements before the airplane is constructed. Changes are made until everything is as satisfactory as accompanying conditions permit. The final solution is then drawn up and incorporated into the final design.

General Requirements

The cockpit for an airplane should be constructed to give the maximum possible comfort, adequate vision and accessibility to all

controls. In a cabin plane, consideration must be given to headroom, ventilation and vision.

The cockpit and primary control units, excluding cables and control rods, should be so located with respect to the propellers that the pilot or controls are not in the region between the plane of rotation of any propeller and the surface generated by a line passing through the center of the propeller hub and making an angle of 5 degrees forward or aft of the plane of rotation of the propeller.

Unless the airplane carries less than five passengers, the pilot or pilots are located in a compartment separated from the cabin. Entrance to the cockpit from the cabin is permitted, but passage through the cockpit should not be considered as an emergency exit for the passengers.

When the pilot is entirely separated from the passengers, suitable means for communication between the pilot or pilots and the passengers should be provided.

The Department of Commerce requires, at the present time, that an oxygen supply or a supercharged cabin be provided for the crew and the passengers above 18,000 feet altitude, or when operations are for more than 15 minutes above an altitude of 14,000 feet.

The arranging of controls, provision for adequate vision, seating arrangements as well as numerous other problems are best solved by constructing a mock-up in which every item is represented.

Windshields

Unobstructed forward vision for the pilot under all conditions is most desirable. For landing, it should be possible to see the wheels at the moment of contact for easy landings.

Windshields should be so installed that they may be cleaned or opened easily in flight, and arranged further so that the air stream and snow or rain are deflected across the opening, unless the windshields are such that rain, snow or ice will not stick to the surface to impede vision.

Various attempts have been made by both N. A. C. A. and N. P. L. to design a windshield to permit visibility through it even under adverse conditions. A few likely windshield arrangements are shown in Figures 10A, B, C, and D.

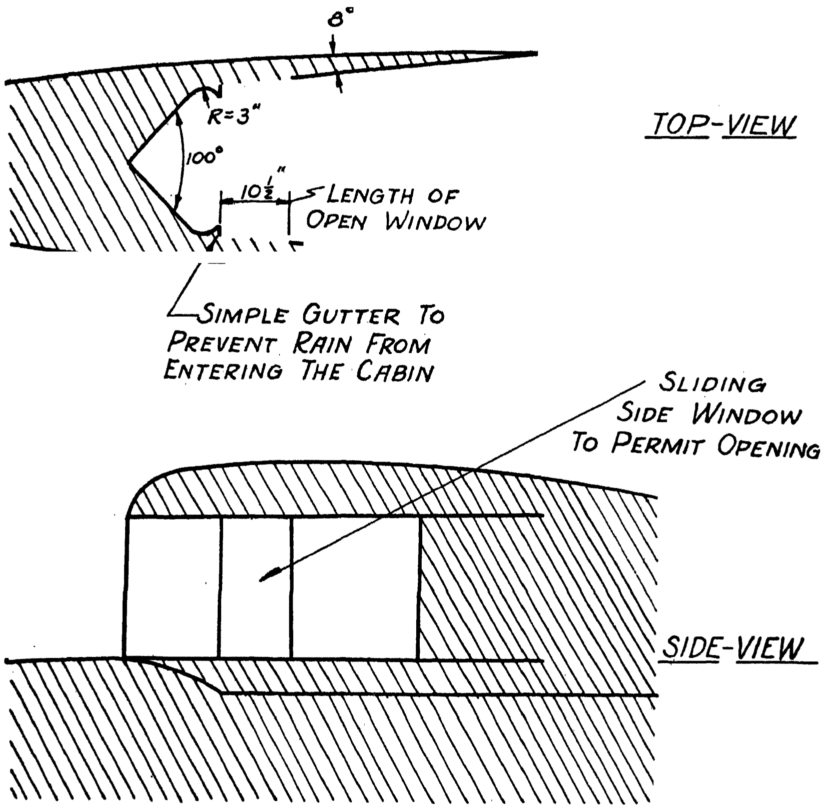
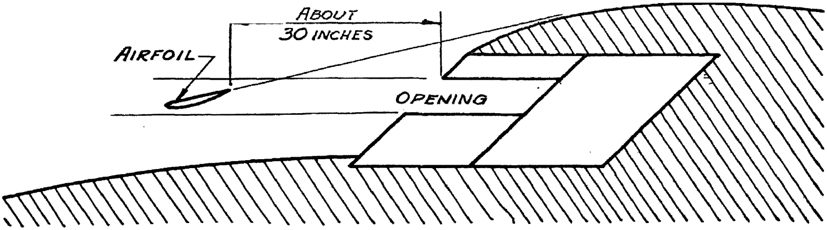
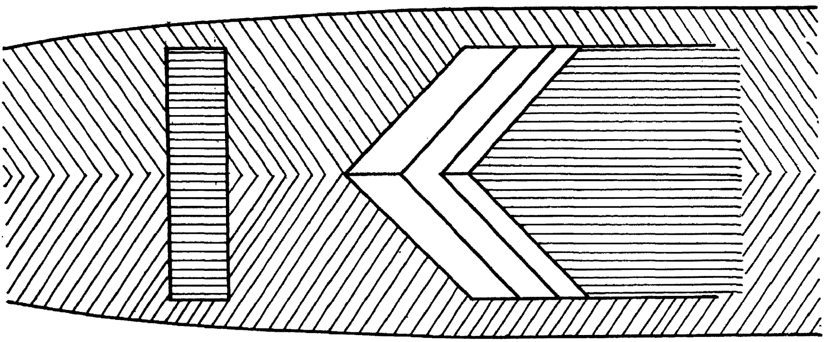


FIGURE 10 A

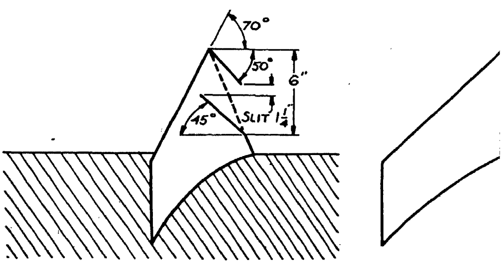


SIDE-VIEW



TOP-VIEW

FIGURE 10 B



SIDE-VIEW FRONT-VIEW

FIGURE 10 C

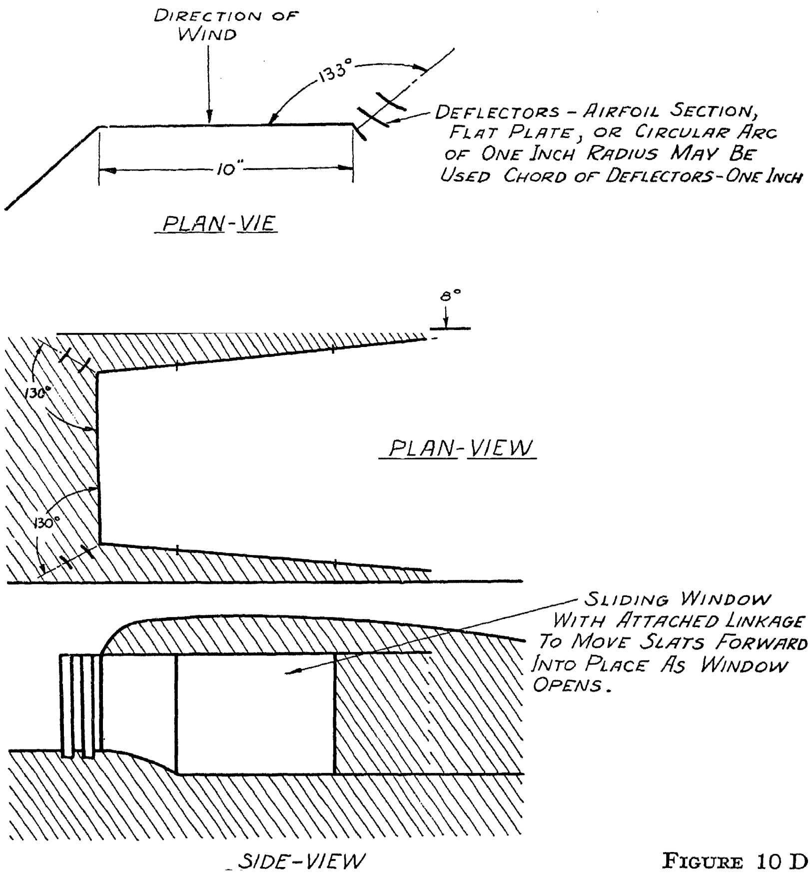


FIGURE 10 D

The windshield arrangement should be such as to give minimum interior reflections and glare which might interfere with the pilot's vision, particularly at night.

In a mock-up the windows may be represented by mirrors which show the parts of the cockpit which may be reflected. By suitable rearrangement of the windshield, these areas may be avoided or provision made to reduce the reflection.

Glare may be investigated by means of an outside source of light which can be moved with respect to the cockpit.

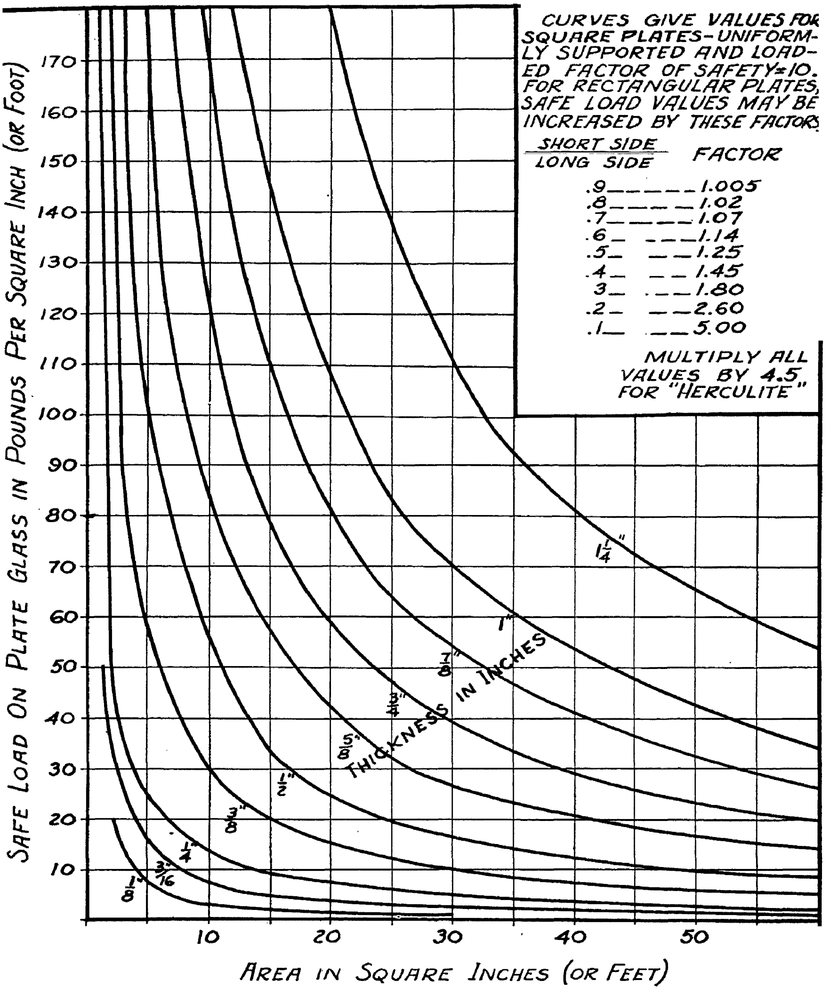


FIGURE 11. VARIATION OF GLASS THICKNESS WITH SIZE AND LOAD

By means of Figure 11, the thickness of the glass for the windshield may be determined. The safe pressure or load that may be sustained by the glass may be calculated approximately by means of the formula $P = 21000\ t^2/Af$ where

P = unit pressure in pounds per square inch (or square foot)
t = area in square inches (or square feet)
A = thickness of glass in inches
f = safety factor, assumed equal to 10

TABLE 5

Characteristics of Glass

Coefficient of expansion of plate glass between —70°F. to ÷ 100°F. is approximately 0.00000823 per degree Centigrade or 0.00000451 per degree Fahrenheit	
Compression	36,000 lb. per sq. in.
Tension	6,500 lb. per sq. in.
Modulus of rupture	3,500 lb. per sq. in.
Modulus of elasticity (Herculite, a tempered laminated safety glass)	10-11,000,000
Weight (¼ inch thickness)	3.29 lb. per sq. ft.

All the windows in the windshield should be made to open quickly, and be removable, if desired. The glass should not lie too flat. An angle between 0 and 45 degrees between the vertical and plane of the glass when the airplane is horizontal is recommended. If the glass lies too flat distortion of vision or undesirable reflections from the sky above may result.

The thickness of the glass depends upon the type and size of plane.

In general, non-shatterable glass, at least $\frac{3}{16}$ inch thick should be used. Anything thinner is subject to accidental breakage or warping.

Seating

The pilot should be comfortably seated. His seat should be adjustable vertically, as well as fore and aft, and angularly. The back of the seat should not be too high.

The seat should be provided with arms, so designed that the inside arm can be swung out of the way when getting in or out of the seat.

When a wheel control is used, the height of the wheel should be such that it will clear the pilot's legs with the seat in its highest position so that the range of seat adjustment will not be limited.

Typical installations of the pilot's cockpit are shown in Figures 12 and 13.

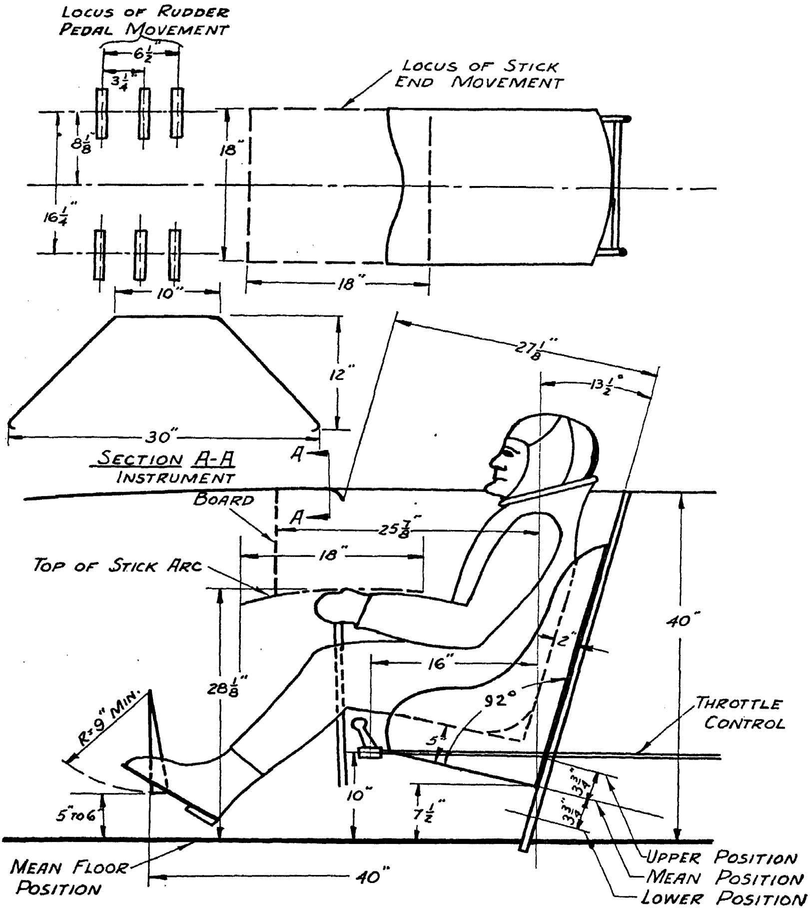


FIGURE 12. OPEN COCKPIT DIMENSIONS. ARRANGEMENT MAY ALSO BE USED FOR CLOSED COCKPITS

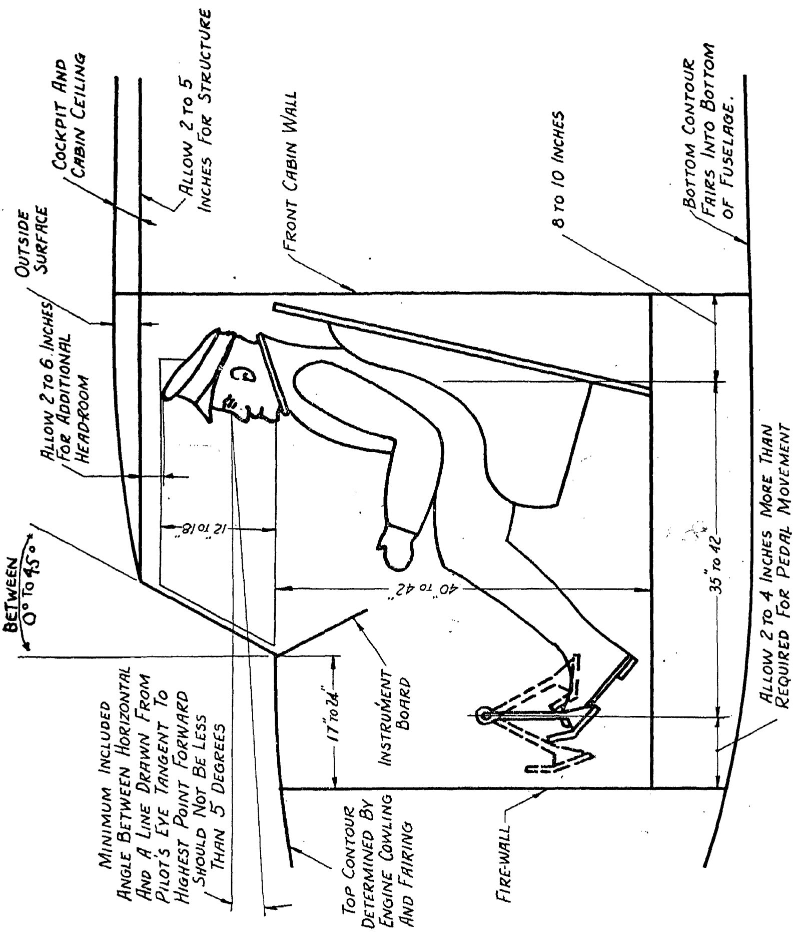
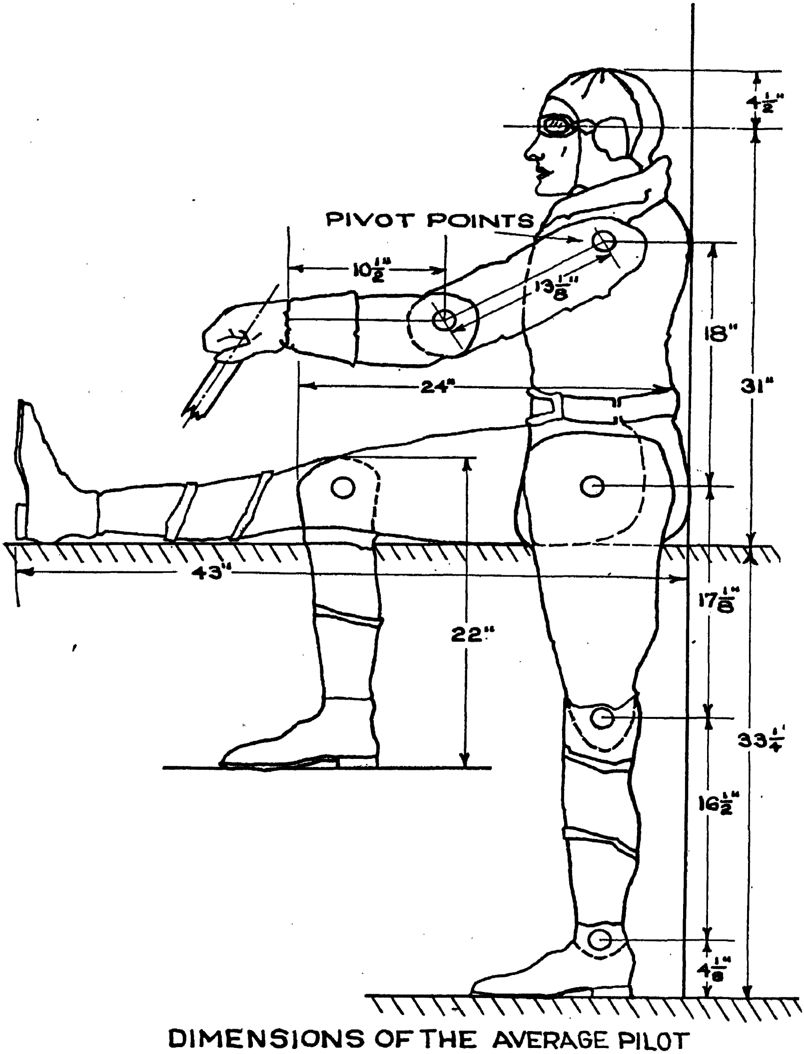


FIGURE 13. GUIDING DIMENSIONS FOR AN ENCLOSED COCKPIT

Figure 14 gives the average dimensions for the pilot. It is desirable to make a celluloid model to scale, so that all cockpit dimensions can be easily checked.



PIVOT POINTS
10½"
13⅛"
18"
31"
24"
4½"
43"
22"
17⅛"
33¼"
16½"
4⅛"
DIMENSIONS OF THE AVERAGE PILOT

FIGURE 14

Exits

If the cockpit is not accessible from the cabin, a separate door should be provided for the cockpit. It is generally desirable to have an emergency exit as well. If the windshield is large enough when opened, it may serve as an emergency exit; otherwise one should be provided.

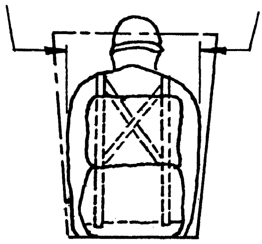
When the airplane is not equipped to carry passengers, the pilot's compartment may be either open or equipped with a hatch of such size that it can be used for an emergency exit for the crew with parachutes.

An opening for an emergency exit should be at least 17 x 24 inches when rectangular in shape, or 24 inches in diameter if circular. Such openings are generally, however, not large enough for a man equipped with a parachute, and are usually intended for emergency exit on the ground.

Parachutes

It is not customary to provide the crew with parachutes when passengers are carried. When the airplane does not carry passengers, pilots should be equipped with parachutes. Figures 15A, B, and C show seating provisions to be made for different type parachutes.

DISTANCE BETWEEN SIDES OF SEATS
MUST NOT BE LESS THAN 18 INCHES.
SIDES MUST BE WITHOUT PROJECTIONS
CONTINUOUSLY FROM SEAT TO TOP OF FUSELAGE.



A.

FIGURE 15 A

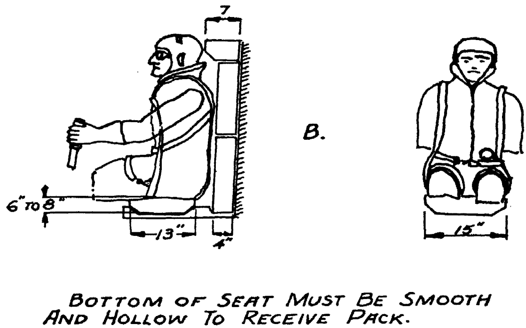


FIGURE 15 B

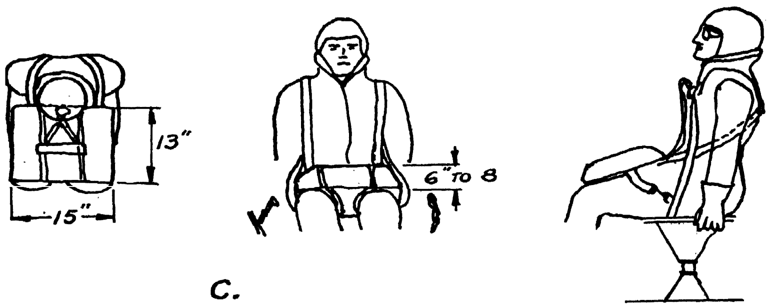


FIGURE 15 C

Controls

All controls should be readily accessible.

The distance from a control wheel in its rearmost position to the back of the pilot should be at least 12 inches. The height of the wheel should clear the pilot's legs with the seat in its highest position.

The maximum movement of the controls should correspond to the maximum movement of the control surfaces.

In a small airplane, seating two to five passengers for example, it is sometimes customary to place a passenger alongside the pilot. In such a case, the control column or stick should not be located between the pilot or passenger unless the "throw-over" type of wheel control is incorporated.

The controls should be so arranged or constructed that neither the passengers, crew nor cargo will interfere with the operation of such controls during the course of flight of the airplane. Further, all controls must be so located and designed that the pilot or co-pilot will not bump them when moving in or out of his seat, or in and around the cockpit.

Instrument Board

There is no agreement as to the best arrangement of the instrument board. It is customary to place the automatic pilot in the top center of the instrument board, the flight instruments and power plant instruments in front of the pilot. The controls and switches should be readily accessible, and in case of pilot and co-pilot arrangements, the engine controls should be located between the two.

In a twin engine airplane, at the present time, there are some sixteen controls for the power plant alone; about eleven flight control instruments in addition to the flying controls, landing gear retraction, lights, heating and ventilating control, radio, brakes as well as flaps and tab controls, for which provision on the instrument board or in the cockpit must be made.

The instrument board should have sufficient room behind it to provide easy access even to the largest instrument.

The instrument board may be made of any material, although aluminum alloy is commonly used.

Drawings Required

The installation drawing of the pilot's cockpit shows the side view of the cockpit and should be drawn to a scale of $1\frac{1}{2}$ inches = one foot.

The drawing should include

1. Cockpit outline
2. Windshield outline
3. Chair details
4. Control stick or wheel placement and movement
5. Rudder control
6. Brake control
7. Flap, stabilizer, etc., control
8. Instrument board location
9. Probable angles of vision
10. Power plant connections
11. Miscellaneous items

A top view is also necessary to show side by side seating, width of seats, clearances and windshield details.

VI. Instruments

From the first flight of the Wright brothers, and practically up to about 1928, pilots flew by feel and intuition, but today they cannot trust their own senses at high altitudes or in fogs or in cross-country flights or in blind flying, since the airplanes are becoming faster and more complicated with each new design. Fogs, high altitudes and night flying obliterate landmarks. High headwinds, sleet, snow and rain interfere with the intuitive senses of the pilots. They must rely, for safety's sake, almost entirely on radio communication, radio beacons, range compass findings, gyroscopic compasses, automatic pilots, turn and bank indicators, and at least twenty-five or more other dials and gadgets essential to the safe operation of the airplane in all kinds of weather.

In transoceanic flying the job becomes even more complicated. The captain of a transoceanic flying boat must be also a first class seaman, a master mariner, a radio expert, an aeronautical engineer, an engine and airplane mechanic, a celestial navigator and a host of other things. But no matter how well qualified the pilot may be along these lines, he would be entirely helpless unless there were instruments or equipment to show how the engines were performing, what speed the airplane was making, at what altitude he was flying (so as to clear mountains), or what course he was following and the innumerable other indicators required.

This wide variety of information has to be given by reliable instruments, and due to the diverse nature of information, the instruments, gadgets and mechanical aids become so numerous that the ideal arrangement of the various instruments, for example, on the aircraft instrument board has been the subject of considerable controversy among pilots since the pilots' needs also vary with the type of flying he has to do. To this day, it has been impossible for pilots to agree upon a definite location of even a few instruments. This state of affairs may be better understood when it is considered that the use of various instruments differs considerably for each particular flying problem; certain instruments, for example, necessary to make instrument landings (i.e. making landings aided by instruments only and not by actual view of the airport on which landing is taking place) are rarely, if ever, used in normal landing. The best instrument board, therefore, is the best compromise that has

been worked out for the particular conditions the airplane has to meet.

Of course there are difficulties. It would be highly desirable for a pilot to be able to step into a strange airplane and feel reasonably sure that he will be able to locate any particular instrument in the usual place on the instrument board. Such standardization, in addition to the reasons already pointed out, is also difficult due to space and structural limitations, as well as due to the size and shape of various airplanes. Some degree of standardization is possible if the majority of the more important instruments are placed in the same general location on the panel.

The military services prescribe some specifications, and the various airlines are wont to set forth their own. Manufacturers of aircraft are able to follow these in most instances except for the relatively small airplane where some elasticity is necessary because of the small area of the instrument board usually available.

The standardization, so far made, calls for the following arrangement. The primary flight group is immediately in front of the pilot and near the top of the panel. This group consists of the Sperry turn indicator and the Sperry flight indicator both on the same level. Below these is the secondary flight group consisting of the airspeed, bank and turn indicator combined, and the rate of climb instruments. To the left, either in the same row or as close as possible, the sensitive altimeter is located. In addition it is customary to locate the magnetic and radio compasses as conveniently close to the other flight instruments as possible.

The engine instruments are usually grouped in the same general pattern, depending upon their number, with the tachometer as close as possible to the flight instruments.

There is often a series of other instruments placed according to the best location available as well as the particular purpose for which the instruments were designed.

Instrument Board

The instrument board may be of any convenient material although aluminum or aluminum alloy is most commonly used. For small instrument panels, plywood may be used. Instrument boards

of 17ST or 24ST aluminum alloy sheet are made in thicknesses varying from $\frac{1}{16}$ to $\frac{1}{8}$ inch. Formica or Bakelite instrument boards are from $\frac{1}{8}$ to $\frac{1}{4}$ inch thick.

The instrument board should be fully accessible from the rear so that any instrument may be readily removed. Special attention should be paid to the depth of the instrument, space required for removal, and extra space for access to connections and the like.

Since instruments should not be subjected to vibration, the board should be mounted on either felt or sponge rubber. Elastic stop-nuts are excellent for fastening down bolts which do not need extra lock-washers. As the board with the instruments has sufficient weight of its own to acquire a vigorous vibration, both it and the individual instruments should be mounted carefully with respect to vibration. The compass particularly should be as independently mounted as possible.

Location

Instruments should be located far enough away from the pilot to permit him to see all the instruments at a glance. If there are a great many instruments, the more important ones should be so grouped that they may be seen readily.

The instrument board location is usually determined by cockpit limitations, control stick clearances and the like.

Grouping

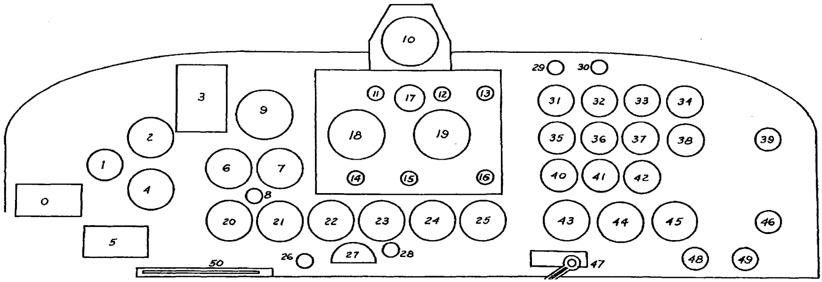
The grouping of instruments depends upon the functions they perform.

Instruments relating to engine operation should usually be close together and preferably right in the line of sight of the pilot; likewise instruments used for navigation, instruments used for communications, and instruments and accessories used for heating and ventilating and similar occasional operational functions should be grouped together. Authorities differ as to the desirable arrangement of instruments in a particular group.

Arrangements of instruments of a typical installation are shown in Figures 16, 17, 18 and 19.

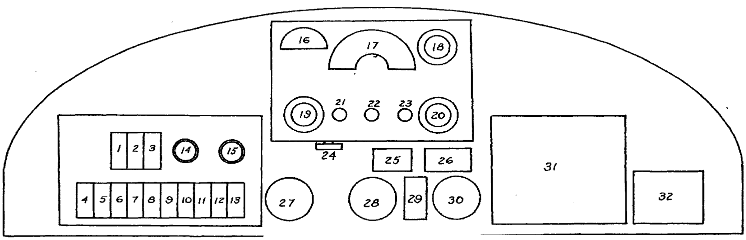
Variety of Instruments

The variety of instruments that may be included in the modern airplane can best be realized from the following incomplete list.



0-Keys, Documents, Etc.	18-Directional Gyro	35-36-Carburetor Temperature Indicators
1-Oil Pressure Gage For Automatic Pilot	19-Bank & Climb Indicator	37-38-Engine Oil Temperature Gages
2-Clock	20-21-Engine Tachometers	39-Hydraulic Brake Lever
3-Department of Commerce Placard	22-Airspeed Indicator	40-41-42-Fuel Tank Gages
4-Measures Air And Gas Mixtures	23-Bank & Turn Indicator	43-Ice Warning Indicator
5-Sensitivity Control For Automatic Pilot	24-Altimeter	44-45-Engine Temperature Gages
6-Manifold Pressure Gage	25-Rate of Climb Indicator	46-Primer For Engines
7-Manifold Pressure Gage	26-Cabin Air Control	47-Vacuum Pump Selector
8-Pull To Clean Supercharger Lines	27-Vacuum Selector Valve	48-49-Fuel Pressure Warning Light Switches
9-Sperry Artificial Horizon	28-Carburetor Heat Control	50-Wing Flap Operating Lever
10-Magnetic Compass	29-30-Fuel Pressure Warning Lights	
11-16-Controls For Automatic Pilot	31-32-Engine Fuel Pressure Gages	
17-Vacuum Gage For Automatic Pilot	33-34- " Oil " "	

Figure 16. The Main Instrument Board



1-Electrically Operated De-Icing Valve Switch	19-Volume Control For Radio Range Beacon
2-3-Switches For Left & Right Landing Lights	20 " " " Two Way Radio
4-Hostess Signal	21-Radio Control Switch
5-Switch For Passenger Instruction Sign When Landing	22-Dial Light Switch
6 " " Cabin Heating System	23-Automatic Volume Control Switch
7 " " Cabin Lighting System	24-Radio Range Beacon Tuning Dial
8 " " Pitot Heater	25-Switch For Electrical Engine Instruments
9 " " Red Nose Headlight	26-Switch To Start Either Motor
10 " " Navigation Lights	27-Ignition Switches For One or Both Engines
11 " " Instrument Panel Illumination	28-Electrical Engine Synchronizer Indicator
12 " " Compass Illumination	29-Switch For Generator
13 " " Flying Instrument Illumination	30-Combined Volt-Ammeter
14-Rheostat For Compass & Directional Gyro Lights	31-Fuse Box
15 " " Instrument Panel Lights	32-Retractible Landing Gear Warning Indicator
16-Gage For Measuring Beacon Volume	
17-Beacon Frequency Selector	
18-Frequency Selector For Second Radio Set	

Figure 17. A Subsidiary Instrument Board

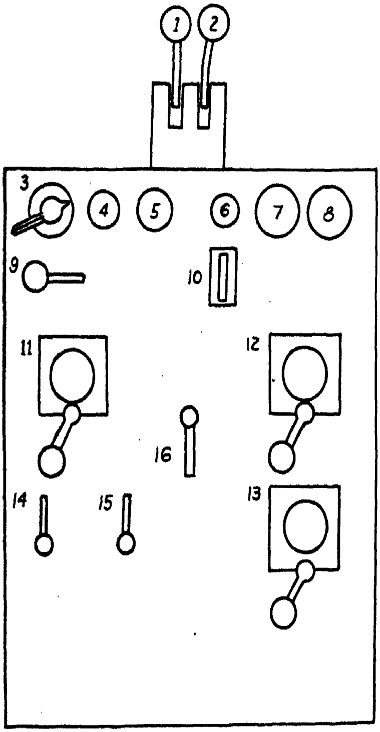


FIGURE 18. CONTROL BOARD USUALLY LOCATED BETWEEN THE PILOT AND THE CO-PILOT

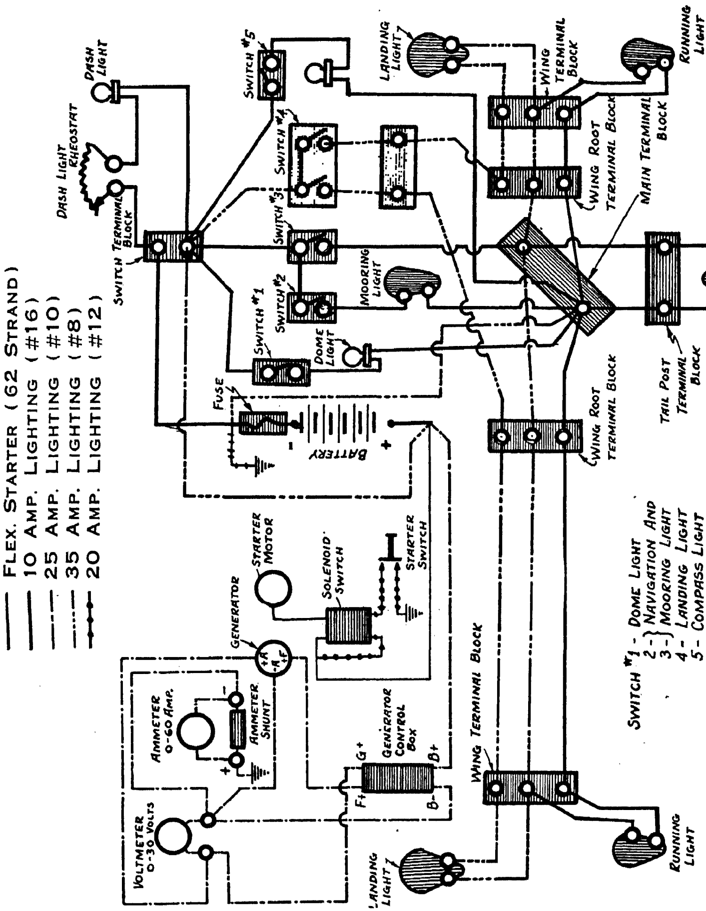


FIGURE 19. ELECTRICAL DIAGRAM OF THE LIGHTING SYSTEM

1. *Air Distance Recorder*—records the number of air miles.

2. *Air Speed Indicator*—registers the actual air speed, referred to standard air, at all times.

3. *Altimeter*—a visual registering but not recording barometer, except that instead of indicating the atmospheric pressure in equivalent inches of mercury, the dial is graduated to read the altitude in feet corresponding to the pressure.

4. *Accelerometer*—may not be incorporated in the instrument board layout. It is used to determine the maximum loads imposed upon the airplane in gusts or in landing. It is often carried to give research workers additional data, but is not required or used by the pilot for his information.

5. *Ammeters and Voltmeters*—these instruments measure the power used for various instruments or equipment which require electrical power such as radios, telephones, electrically operated retraction gear and the like.

6. *Automatic Pilot*—a device manufactured by the Sperry Company used to operate the control surfaces to maintain an even keel without the intervention of the pilot. The automatic pilot relieves the pilot of much strain of constantly "flying" the airplane so that he can devote more attention to the behavior of the power plant and other equipment.

7. *Bank Indicator*—a visual indicating device for registering the angle of bank.

8. *Climb Indicator*—registers the rate of climb in feet per minute, so that the pilot has a ready means of determining whether he is climbing fast enough. If it registers zero, the airplane is flying level; if it registers negatively, the airplane is descending. The dial is marked to read "Up" and "Down."

9. *Clock*—the clock is very important for navigation and maintaining schedules.

10. *Compass*—there are several types of compasses—magnetic and radio. Both are usually used for cross-checking purposes. For private airplanes, a compass is required when the range is 100 or more miles, or when operating over large bodies of water beyond sight of land.

11. *Drift Indicator*—an instrument which measures the angle between the actual flight path of the airplane and the longitudinal axis of the airplane (which indicates the direction in which the airplane is headed).

12. *Fuel Level Gauge*—a long distance indicating gauge for registering the actual fuel level in the fuel tank. If there are several fuel tanks, a separate gauge for each tank is required.

13. *Fuel Pressure Gauge*—indicates the fuel pressure in the fuel line—necessary for determining whether the fuel is brought to the engine in the proper quantities.

14. *Flap Control*—either a manually-operated device for controlling the flap or an electric switch for starting electric control. It may also be a hydraulic pump handle.

15. *Generators*—attached to the engine at locations provided for them. These generators generate the power required for lighting and for motor-driven appliances.

16. *Manifold Pressure Gauge*—registers the actual pressures in the engine manifolds. Nowadays a close check is kept on the manifold pressures which are changed from time to time by the pilot to give the necessary power for take-off and cruising.

17. *Octant*—used for celestial navigation. The instrument measures the vertical angle between the horizon and a celestial body.

18. *Oxygen Equipment*—this equipment consists of various regulators and indicators as well as tanks and distributing units. Airplanes flying above 14,000 feet any great length of time are required to carry this equipment.

19. *Propeller Controls and Indicators*—propellers of today are usually of the controllable pitch; whether automatic, manual, or semi-automatic; there are certain controls and pitch indicators present. These vary according to the type of propeller.

20. *Pumps*—there are fuel pumps either hand operated or engine driven; there may be pumps for the hydraulic systems, and in some cases, oil pumps may also be required. Pumps operated by hand, or used for intermittent duty only, are usually operated from the instrument board.

21. *Radio*—safe flying is dependent upon radio and telephone communication. This equipment has a number of switches, dials and controls which must be so located as to be easily accessible to the co-pilot or radio man.

22. *Suction Gauges*—suction gauges indicate the suction pressure in the lines leading to the automatic pilot and other instruments depending upon suction for their operation.

23. *Sperry Artificial Horizon*—a visual indicating device used to indicate the "horizon" when the actual horizon is obscured. It is of particular importance in all blind flying operations when the intuitive sense of man is decidedly unreliable.

24. *Switches*—ignition switches, radio switches, light switches, electrical appliance and equipment switches in any number may be found in the cockpit.

25. *Starters*—control for the operation of the engine starter may be found in the cockpit unless the airplane is small.

26. *Tachometer*—it is very important that the crankshaft speeds of the engines be known at all times. Any sudden change is a warning that something must be wrong. The revolutions per minute indicated by the tachometer also serve as some indication of the actual power developed by the engines on take-off; the indicated r.p.m. is high since more power is required for the take-off than at any other time. During cruising, the r.p.m. of the engine is lower than for high speed since less power is required and better fuel and oil consumptions are obtained at lower engine speeds.

27. *Thermometers*—these are used to register oil temperatures; to register the temperature of the coolant in case of liquid-cooled engines; to register cabin temperatures for heating and ventilating purposes; to register the outside temperatures—especially important when icing conditions are likely to be present.

28. *Turn Indicator*—often combined with the bank indicator; when the two are used in conjunction with each other they are considered among the most important flight control instruments, and should be located on the instrument board in the direct and unobstructed forward view of the pilot in the center of what is called the "primary flight group" consisting of the airspeed indicator, the turn-and-bank and the rate-of-climb indicator.

29. *Warning Units*—warning units may be special lights that flash on and off; or buzzing signals; or any similar devices. Very often the positions of the hands on the various indicating dials are sufficient warning signals. Some instruments are so arranged, since the dials may be rotated, that during the most common flying regime, such as cruising, dial hands or pointers of all the most important instruments point in the same general direction. If one is out of alignment, it may be an indication that either the instrument is not functioning or that the particular piece of apparatus with which it is used is not functioning properly—both cases are likely to be present.

VII. The Passenger Cabin

It may seem to be an easy matter to design the passenger cabin, to provide proper seating arrangement, heating and ventilating and vision outward. After all there are only a few rows of chairs placed rather closely together and usually a window beside each chair. But as in all parts of the airplane, numerous considerations enter into the final solution which is the result of many compromises of arrangement, structure and balance, as well as weight.

From the passenger's viewpoint, as much room as possible is desirable, but if this consideration were given full weight the size of the cabin might become unwieldy for the type of airplane to be built. Moreover, if all the seats are not occupied, the center of gravity will not coincide with its position for the fully loaded condition, and just as soon as the center of gravity has moved too far from a given position, the pilot will find it difficult to balance, and so to fly, the airplane. This has been discussed earlier to some extent, when the balanced diagram was considered.

Therefore, to avoid a too-large cabin, it is best to start with the question: "What is the minimum allowance for spacing of seats, for aisle widths and for headroom?" A student measured the distance between backs of two church pews, one behind the other, and found it to be 28 inches, but also reported that the spacing was not conducive to comfort. However, one may say that distance is the absolute minimum spacing possible. Any additional allowance adds to the comfort, but discretion must be used. If there is a central fuselage, with the engines located in the wings, greater leeway may be permitted for both fully loaded and empty conditions and still good balance obtained, because the movable or variable load may be placed with its center of gravity coinciding with the center of gravity of the complete airplane. For a single engine airplane, with the engine in the nose of the airplane, it is not usually possible to get this happy solution and so less allowance must be given to the spacing of seats than absolute comfort may require.

There should be no obstructions overhead or on the floor. Sometimes, to avoid too deep a fuselage, a designer will locate the wing spars so that they extend above the floor level. Usually, in such a case, there is a minimum of headroom so that the passenger is likely to stoop to avoid hitting his head and while worrying about that mat-

ter, he is more than likely to forget about the floor obstructions. If floor obstructions are necessary—and that is debatable—a wall with a door in the center may be a good solution since passengers usually expect to step high over a sill, especially if in some way their attention is drawn to it.

Obstructions—even on the side walls—should be prohibited unless the cabin is large enough to permit incorporation of compartments so that side wall obstructions can be hidden in the dividing partitions.

The interior treatment of the cabin often may give an impression of spaciousness without being actually large. White or light-colored ceilings give an impression of height, so that the cabin may be made narrower or shorter without affecting the passengers unfavorably. There is at present much interest in interior decoration of aircraft cabins, particularly in the psychological effects of various treatments.

The major factors affecting the comfort of passengers are:

1. Roominess, which has to be provided through proper seating; as large cubical capacity per passenger as possible; and a reasonable amount of leg room, elbow room and headroom.

2. Proper heating and ventilating, which is the subject of a subsequent chapter.

3. Soundproofing which is of great importance even for airplanes flying for relatively short periods of time, and much more so for transcontinental and transoceanic airplanes which are in the air for many hours.

4. Vision outward so that the passengers may see the country over which they are flying. Although on long flights the passengers may find the scenery monotonous after a short while, there is a soothing effect produced by being able to see out.

5. Conveniences such as toilet facilities, refreshments, reading material, writing desks and other small items which make traveling more appealing.

The notes that follow will help the designer in planning the cabin layout.

Cabin Dimensions

The cabin dimensions depend upon the type of chair, the number of passengers and the constructor's conceptions of comfort requirements.

Two rows of chairs with a center aisle are common practice. Two seats on one side and one on the other helps to concentrate the variable load and is therefore better for stability and trim for empty and full conditions. The latter arrangement requires a wider fuselage which may affect the performance of the airplane by increasing the parasite resistance.

The main requirements for a comfortable cabin are:

1. An aisle wide enough to permit passengers to walk up and down without disturbing passengers in seats. Aisle widths vary from 12 to 24 inches with an average width of 16 inches.

2. An aisle ceiling high enough to permit a tall man with a hat on to walk its length without stooping. Aisle heights vary from 51 inches for the minimum value up to 75 inches maximum. Except for possible private use, a passenger-carrying airplane should have an aisle height of at least 72 inches. Elsewhere it is not necessary to have such height as to permit standing erect, but the use of members over the seats that may be struck by a passenger rising should be avoided.

3. A minimum width for a cabin seating two passengers should be 66 inches with an additional 25 inches for each additional passenger across width of plane. Cabin widths for two rows of passengers vary from 52 inches for small cabins up to 66 inches for cabins seating 14 or more passengers.

A wide aisle gives the impression of spaciousness and may permit some reduction in the allowance for headroom.

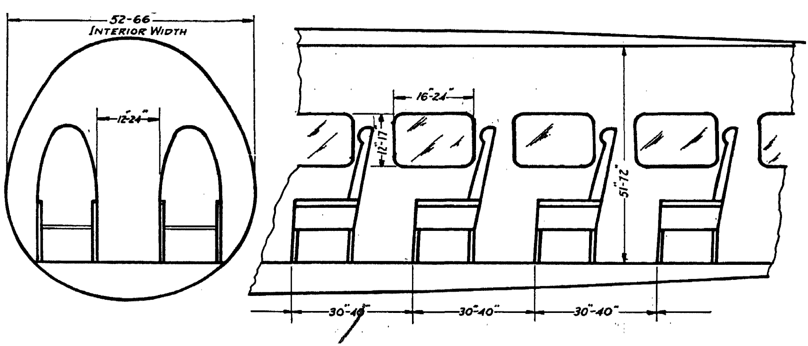


FIGURE 20. SUGGESTED SPACING FOR PASSENGER SEATS

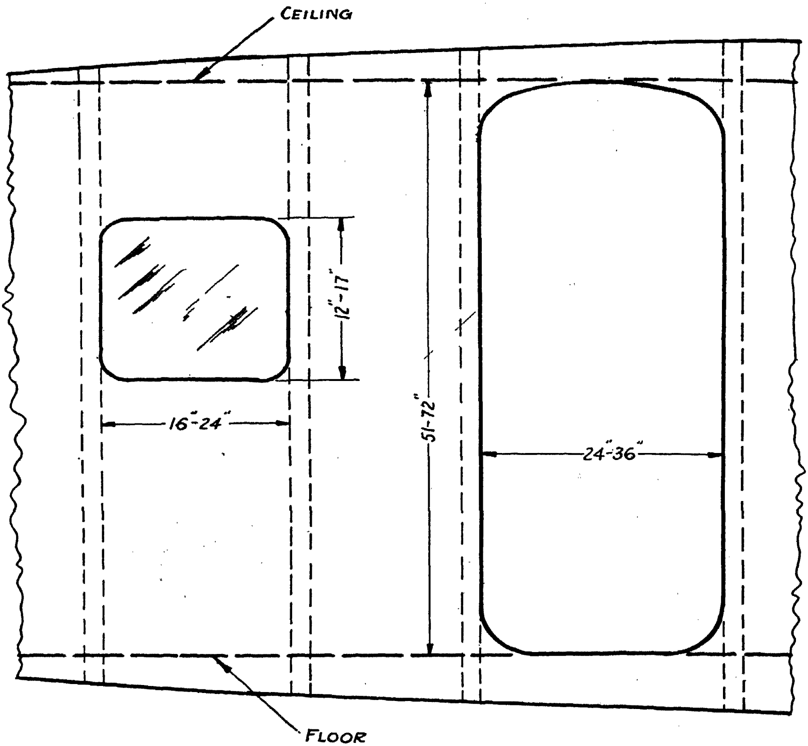


FIGURE 21. RECOMMENDED SIZES FOR WINDOWS AND DOORS OF A PASSENGER CABIN

The cabin should be free from structural members. If these are unavoidable, separate compartments provided for groups of passengers may overcome this objection.

Passenger Seats

Passenger seats should incorporate the following features:

1. Seats should be adjustable with respect to the angle of the seat with the floor and the angle of the back of the seat to the vertical.

2. Seats should be reversible to permit the passengers to face each other if they care to.

3. Seat adjustments should be made easily and quickly by the passenger.

4. A foot rest should be provided in the form of an apron or shelf, preferably adjustable, attached to the seat ahead.

5. Seats should be permitted to swivel for easy egress if cabin clearances and weight allowances permit. Placing the chair at a slight angle to the fore and aft axis of the cabin accomplishes practically the same result.

6. Easy egress may be obtained also if the outside arm of the seat can be dropped.

7. Cushions on seat bottom should be deep and provided with springs as well as padding to reduce effects of bumps on landing and in the air.

8. Direct metallic contact of the seat with the structure may be avoided by means of suitable rubber shock-absorbing connections.

9. Width of each seat should be at least 24 inches, with one inch clearance between seat and inner lining of the cabin. A minimum width of 19 inches may be permitted.

10. For a narrow cabin, the backs of the seats should be tapered toward the top, especially those paralleling the aisle. This design permits more room for passenger going down the aisle.

11. Seats should not weigh more than 20 pounds. Seat weights vary from about 12 to 20 pounds.

12. Seats may be made convertible into berths by lowering back to form bottom of berth.

13. Ample leg room should be provided. Seats are spaced from 30 to 40 inches apart, with an average of 36-inch spacing for cabin passenger capacities of less than 12 persons.

14. All passenger seats should be identical.

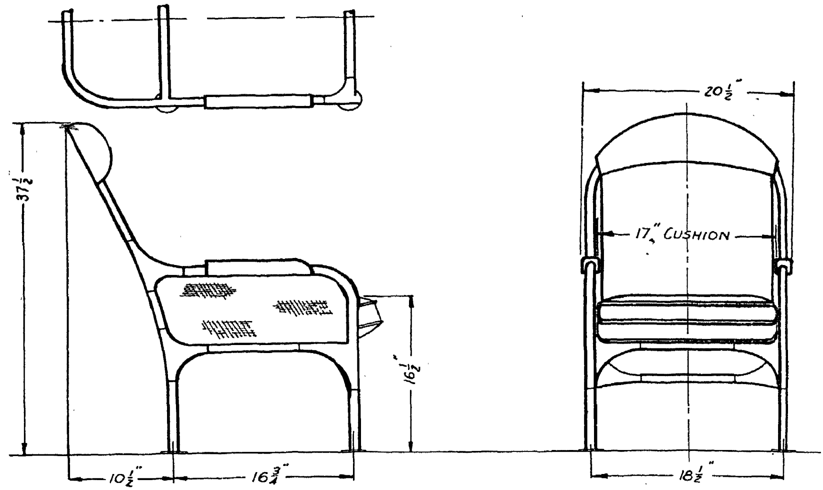


FIGURE 22. SUGGESTED DIMENSIONS FOR A PASSENGER CHAIR

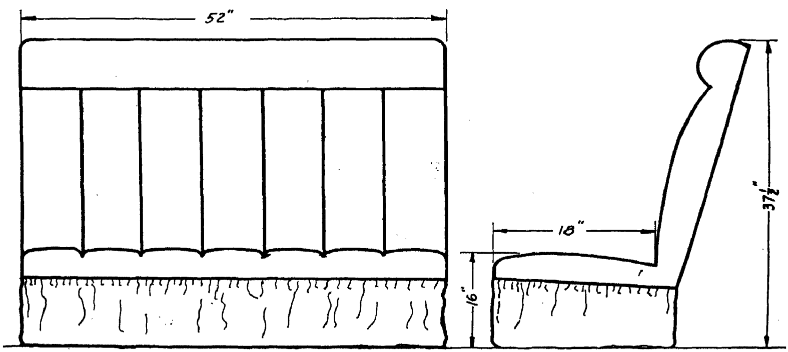


FIGURE 23. SUGGESTED DIMENSIONS FOR "DOUBLE CHAIR"

Sleeping Accommodations

Separate compartments are desirable when sleeping accommodations are to be provided.

Berths vary in width from 28 to 32 inches with 30 inches a good average value. (See Figure 24.) The length of the berth varies from 72 to 76 inches with a preference shown for the higher value.

Mattresses, blankets, pillows and bed linen for a berth weigh from 12 to 18 pounds.

Windows

A window should be located at each seat with the bottom of the window at about shoulder level of the passenger when seated. The window should also be slightly forward, that is, the right side of window as one faces it on the right side of the cabin, for example, should be somewhat ahead of the back of the seat.

The size of the window depends upon the type of structure. The windows vary in size from about 12 inches square or oval up to 17 $\times$ 24 inches, or 24 inches in diameter when circular; if easily removable, they may be classed as emergency exits.

The windows should be recessed in felt-covered rubber window stripping to reduce vibration and noise. They should be designed so that the passenger cannot open them.

The window glass is usually $\frac{3}{16}$ inch thick. Less thickness would be dangerous since a passenger might inadvertently put his hand through the glass. Windshield glass for the pilot's cockpit varies from $\frac{3}{16}$ to $\frac{1}{4}$ inch in thickness. For supercharged cabins for high altitude flying, the windows have to be made considerably thicker.

Lighting

Individual lights usually placed slightly overhead are provided for each passenger; they should be subject to control by the passenger but a master switch should be located in the pilot's cockpit so that the pilot may shut off the light when necessary.

Furnishings

The walls of the cabin are usually treated for soundproofing. The interior may be finished in Fabrikoid, painted and doped aircraft cotton or some suitable upholstering material.

Inspection panels (in the case of fabrics Talon fasteners are suitable) must be provided to permit ready access to the structure underneath for inspection purposes.

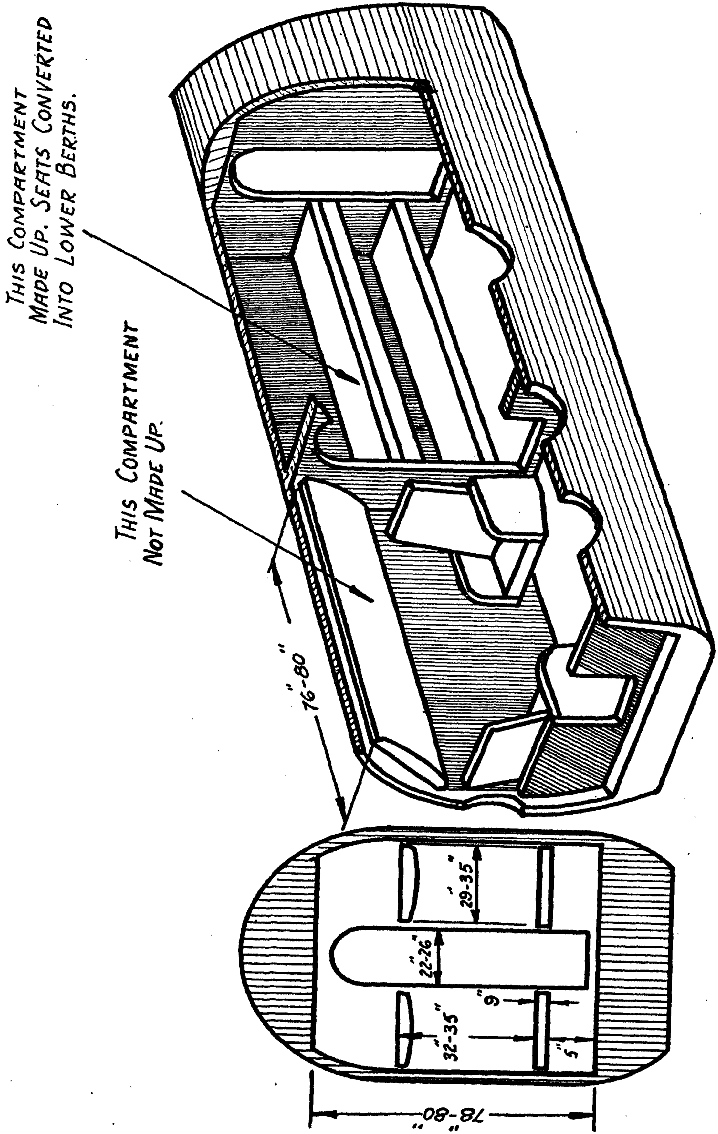


FIGURE 24. THE INTERIOR ARRANGEMENT OF A "SLEEPER" AIRPLANE

Curtains for windows should be installed so as to permit their removal and replacement in one minute or less.

For larger transport airplanes, provision for attachment for small tables for playing bridge or holding a typewriter should be made.

Since many trips are long, any device used to entertain the passengers will be welcome. Airspeed indicators, altimeter and probably a compass located on the front wall of the cabin will draw attention. An illustrated map of the country being traversed will add to the general interest of the trip.

Provision for Airsickness

Paper bags are usually found in a pocket attached to the back of the seat. Wash-room facilities are of course provided. The steward or stewardess is also prepared to take care of such cases.

Flooring

The cabin flooring varies in treatment in various designs. It should be light and soundproof, as well as sound and vibration damping. A free floating flooring made possible by the use of rubber or felt between the floor and the primary structure is desirable.

A thickness of balsa wood between thin face plies of birch or aluminum alloy makes a very effective flooring which will take considerable local loads. The floor may be further treated with a thin cork tile or "battleship" linoleum for appearance and wear.

Floor panels should be provided with means for quick and easy access to the structure below for inspection purposes.

No obstructions, such as spars, should extend above the floor level. It is customary to rest the floor on the spars. Inasmuch as the airplane may take any flight attitude it is difficult to state what constitutes a "level" floor. A good method is to select the angle at which the cruising speed of the airplane occurs and make the floor level for this condition. A few degrees variation will not be serious.

Toilets

Passenger airplanes are equipped with toilet and wash-room facilities, in a compartment separated from, and usually in rear of, the cabin.

The toilet (see Figure 25) consists of a metal container with a water-proof liner of sufficient capacity with a suitable chemical preparation. It should be simply designed and easily cleaned.

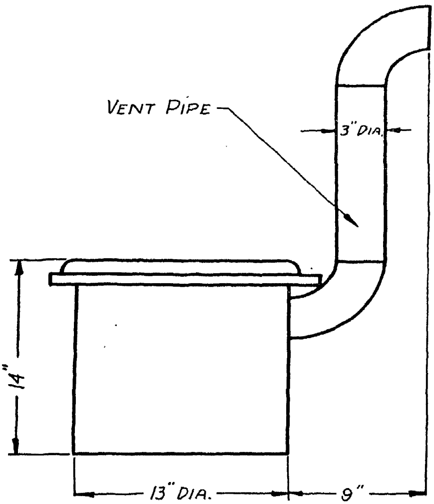


FIGURE 25. DIMENSIONS OF A CHEMICAL TOILET

Wash basins are small but useful. The water supply should be placed in a tank so located that it can be filled readily from the outside, but should be protected against freezing. (Figure 26.)

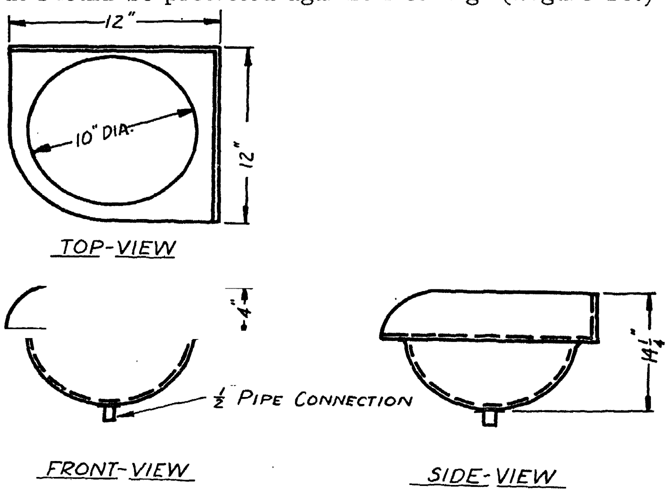


FIGURE 26. A CORNER WASH BASIN

A water allowance of 2 to 3 pounds per passenger, depending upon the range, is ample for both drinking and washing purposes. (Figure 26.)

Refreshments

Since flying schedules do not permit long stops at intermediate landing fields, it is becoming customary to provide passengers with a light lunch, usually made up in boxes. A weight allowance of $\frac{1}{4}$ to $\frac{1}{2}$ pound per person is ample for the food, container, paper napkins, etc. On larger and newer equipment of passenger airlines, a greater allowance may be made to permit furnishing light, hot lunches in flight.

Baggage Compartment

Since only coats and hats which the passengers are wearing are permitted in the cabin, suitable baggage compartments must be provided. These compartments are not accessible in flight.

To expedite the removal of luggage, mail or express, these baggage compartments must be accessible from the outside by means of separate baggage doors.

Usually two or more baggage compartments are provided in the fuselage to permit the limiting of center of gravity movement for the various loading conditions possible. The baggage door carries a schedule giving the load capacities of each baggage compartment and for fully loaded and partially empty conditions of the airplane.

Luggage varies in size so that it is difficult to recommend exact cubical capacities of baggage compartments. A standard suitcase measures $9 \times 17 \times 31$ inches.

Mail compartments should allow $12\frac{1}{2}$ cubic feet for each 200 pounds of mail.

VIII. Heating and Ventilating

The importance of heating in an airplane cabin can be gauged easily when it is realized that the temperature of the outside air decreases rapidly with altitude. When it is some 60°F. on the ground, it may be —50° F. at 30,000 feet. In these days of sub-stratosphere and stratosphere flying, the change in temperature that an airplane encounters from the moment it leaves the ground until it reaches its required altitude a few moments later is enormous, and it should be especially noted that this change takes place in a few minutes.

A difference in temperature may also be experienced as the airplane flies from a warm country to a cold country. While this change may not occur in a few minutes as in the case above, it does occur during the same flight so that a heating system has to be provided whether flying is during the summer months or not.

It is not the heating alone, however, that has to be considered, but the ventilation as well.

It is estimated that a person requires about 25 cubic feet of air per minute. Normal air contains about 21 per cent of oxygen by volume of which a person removes about 4 per cent each time he breathes. Since it is possible to live in an atmosphere containing about 12 per cent oxygen, this original 25 cubic feet of air could be breathed a little more than twice, say three times; thus the air would last about three minutes. If a person were staying in a room about two hours, he would need 1000 cubic feet. Neglecting infiltration, auditoriums would have to be designed to allow approximately 1000 cubic feet for each occupant if they are to remain for 2 hours. It is possible to design auditoriums for such volume, but it certainly is not feasible to design aircraft cabins for any such volumes.

Since the volume available per passenger in an aircraft cabin probably varies from 40 to 60 cubic feet, regardless of the length of time the passenger remains in the cabin, it is necessary to bring air into the cabin constantly to replenish the supply. Some air will filter through seams and cracks, but such air will have the temperature of the outside air which may be very cold. It is necessary therefore to be able to bring air in from one central inlet under control, and to remove it by suitable and positive means.

This change of air every few minutes complicates the heating problem. In a room there are steam radiators which will heat the air, and since the air is removed very slowly by natural infiltration outward through windows, doors and cracks in the walls, it is a comparatively easy matter to maintain a constant temperature. In an aircraft cabin, steam radiators would be useless unless the walls, ceiling and floors were radiators. It is therefore necessary to heat the incoming air before it actually enters the cabin.

In designing the heating system the following factors as explained under the various headings must be considered.

Physical Conditions

In summer, temperatures of greater than 100°F. are seldom encountered even in the warmest climes. Heating and ventilating systems have not yet incorporated systems for cooling the air except the cooling of cabins of aircraft by means of an outside refrigerating system before take-off. This cooling of the cabin disappears within a short time after take-off. By proper ventilation, temperatures above normal room temperatures of 72°F. are not felt.

At the lower temperature limit the heating system must be designed to keep both the passenger cabin and the pilot's cockpit warm at a temperature of 70°F. when the outside temperature is about –20°F. When the airplane is used near the Arctic or in other cold regions, the heating system should be designed for air entering at a temperature of –40°F.

The heating system should be designed to take care of these extremes in temperature within five minutes after the engine is started, for warming up, and must be readily controllable to take care of sudden changes in outside air temperatures encountered either in climbing or in passing from one locality to another.

The incoming air should not be lower in temperature than that maintained in the cabin.

In order to regulate the temperature reasonably it is necessary, at times, to deflect a sufficient amount of the incoming air from entering the cabin by means of a diffusing or butterfly valve operated either by an electrically-driven gear box through a thermostat, or by manual operation. Either method has a certain amount of time lag not possible to overcome.

Water separation must also be provided in the system in case of rain in the incoming air. Snow usually offers no problem since it is vaporized easily in the heater system.

A dust separator is especially desirable when the airplane is to operate from dusty airports.

Physiological and Psychological Considerations

The objects of ventilation are of course

1. to supply oxygen,
2. to remove odors,
3. to remove toxic or poisonous substances,
4. to remove body heat, moisture, and heat from other sources.

At present there are no ready means available for controlling the humidity of the air. It is generally believed that the relative humidity should be about 40 per cent for good health conditions. However the movement of the air has some effect. In still air, the body is enveloped in a layer of moist air so that even a moderate air movement tends to create the effect of a draft.

Care should be taken to avoid possible overheating of the cabin. The general effects of overheating may be summarized as follows:

1. Increased heart beat and blood flow.
2. Increased respiration.
3. Increased sweating with attack of cramps.
4. General lassitude and dizziness.

Air Movement for Comfort

The air movement in the cabin should be such as to avoid the feeling of drafts. To this end it is desirable, although not always possible, to meet the following conditions:

1. Limit the maximum air velocity to about 2 feet per second for persons at rest.
2. High velocity currents of air should not be directed downward on the occupants, nor strike them from behind at about neck level nor come up at about floor level.
3. The air should not be admitted to the cabin below the normal cabin temperature.

The natural movement of the air within the cabin is usually from rear to front. However, for various reasons it is desirable to reverse this flow. Normally this reversal is accomplished easily by locating the exhaust at the top rear and due to the forced draft produced by the incoming air and forced exhausting system usually provided, the air flow can be easily controlled.

The toilet room should be made as airtight as possible, and a very positive system of ventilating should be provided for this room independent of the cabin.

It has been found that cool air is most effective in keeping a passenger comfortable in hot weather if an individual ventilator is located above and in front of the seat.

All incoming air should be controlled by the pilot at its source. Windows should not be used for ventilating purposes.

Air Inlets

Air inlets should be so located as to be free from contamination of oil fumes or other engine gases. Since the ducts should have as few bends as possible to permit the normal air pressure in flight to force the air into the cabin, the location of these inlet ports may offer difficulties.

In a single engine plane, a duct carried between the cowling and the engine ahead of the collector ring has been found satisfactory.

For a central fuselage having no engine in its nose the foremost point on the fuselage is an excellent location for this inlet port.

The leading edge of a wing in the pressure region and within the propeller slipstream, if possible, but well outside the region of possible contamination from oil or exhaust gases, has also been found suitable.

Heat Sources

The exhaust gases of the engine supply the main source for heating the air, whether directly or through intermediate means. In the case of the liquid cooled engines, auxiliary radiators may be used for heating the air. In any case, it is desirable to have a very hot source if a light, efficient and instantaneous system is to be obtained.

Air Requirements

About 25 cubic feet of air per person per minute is the minimum limit. This will help to keep down the carbon dioxide content to a desirable minimum. More air may be allowed for if space and weight tolerances permit.

Air Ducts

The inlet ducts for leading the air to the heater vary in cross-section from 3 to 5 inches in diameter. It must be remembered that all openings of this kind add to the aerodynamic resistance so that

choice of cross-section should be guided by the type of airplane—the larger the craft, the larger these openings may be permitted.

The ducts leading to the heater are made either of aluminum, aluminum alloy or stainless steel of a minimum thickness of 0.025 inch.

The ducts leading away from the heater may be of metal, but usually only for a short distance. The longer lengths of the ducts, after the metal section, may be made of doped aircraft fabric. To prevent too rapid cooling of the air while being transmitted to the cabin, and to aid in soundproofing, the fabric (or metal) ducts are lined with about $\frac{1}{4}$-inch thick felt held in place by a light mesh aluminum alloy screen. The outside diameters of these ducts are about the same as the inlet ducts, but their inside diameters become less when the felt has been applied.

Individual ducts are led to each seat. Since the amount of air leaving each opening should be the same, it may be necessary to vary the opening by inserting obstructions. Once the airflow has been regulated for each opening, no changes need be made.

A grill-work made of fine mesh wire should cover each opening so that nothing may fall in the duct.

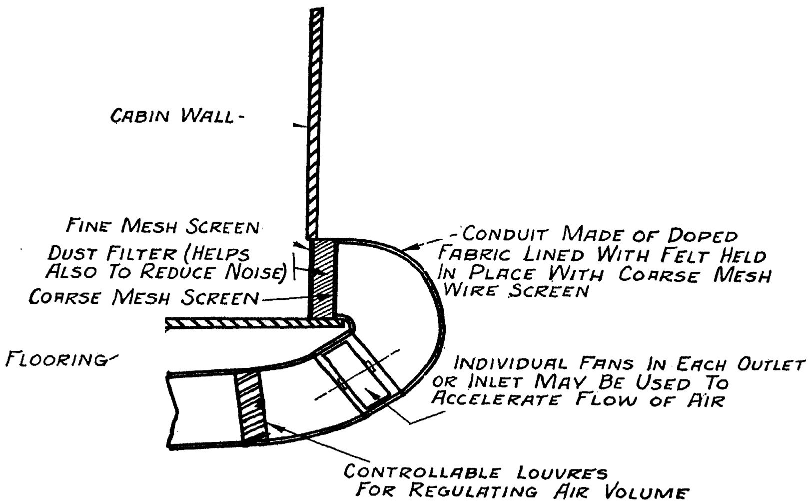


FIGURE 28. CROSS SECTION OF AN INLET AIR DUCT

The outlet ducts should be located along the roof and exhaust near the top rear. In some cases it may be more desirable to exhaust at intervals along the side of the fuselage; however, each additional opening adds to the soundproofing problem.

A separate exhaust system should be provided for the toilet.

The pressure of the incoming air and the slight suction that is usually present on the top surface of the fuselage are enough to force exhaust of the air. If the rate of exhaust is not fast enough, an exhausting fan may be necessary.

Boilers and Radiators

The safest type of system is one in which the exhaust gases heat a liquid which is vaporized or circulated through a radiator. The air is forced through the radiator and then led into the cabin. It is not feasible to supply actual radiators for each pasenger in the cabin since the volume of air brought in must be heated quickly and effectively.

Various systems are illustrated schematically in this chapter.

The liquids used in the boilers may be commercial ethylene glycol which boils at 310°F. to 340°F., specific heat of about 0.70; quenching oils of high boiling points; or a mixture of water and alcohol. The requirements for such liquids are that they boil at about 300°F., do not freeze or congeal at the lowest temperature likely to be encountered and have a flashpoint high enough so as not to offer a fire hazard in case of a leak.

The radiators normally used are the cartridge type, or similar to the oil coolers. It is desirable that the air be in contact with the radiator sufficiently long to be heated, that the heat transfer of the radiator be very rapid, and that the airflow speed be maintained.

Calculations

A typical example will be given here to indicate the type of calculations that may be made.

Temperature of incoming air	—20°F.
Temperature to be maintained	70°F.
Number of passengers to be supplied with air	10
Volume of air per passenger to be supplied	25 cu. ft. per min.
Volume of cabin	400 cu. ft.
Ethylene-glycol for liquid type system.	

The number of changes required will be $\frac{10 \times 25}{400}$ per minute or one change every 1.6 minutes.

Assuming 10 separate outlets one for each passenger for the incoming air, then 25 cu. ft. of air per minute flow through each duct. Restricting the air velocity to 2 feet per second the area, S, of each inlet duct would be

$$2\ S = \frac{25}{60}$$

$$S = \frac{\pi\ d^2}{4} = \frac{25}{120}$$

or $d = .515$ feet approximately—6.18 inches.

Actually the practical diameter of the inlet duct would be 3 inches, with partial obstruction in the opening for regulating the airflow.

The heat balance equation may be written as

$$Q = Q_1 + Q_2 - Q_3$$

where

Q = heat required to raise 250 cubic feet of air per minute from —20° to an average of 70°F. within the cabin.

Q_1 = heat required to raise incoming air to maintain temperature in cabin at 70°, if there were no losses.

$$Q_1 = W\ C_p\ (t_2 - t_1)$$

where

W = weight of air,
C_p = specific heat of air = 0.2375 @ 70°F.,
t_2 = temperature of incoming air,
t_1 = outside temperature.

Q_2 = heat lost by radiation from cabin by windows, fabric, flooring and the like.

Q_3 = heat generated by the passengers. This is usually neglected.

The heat lost by radiation is difficult to ascertain accurately. With well soundproofed and hermetically sealed cabins, this loss may be quite negligible. A loss of 100 BTU per minute will be assumed.

Weight of 250 cubic feet of air at 70°F. = 250 x .076 = 19 lb.
$Q_1 = 19\ (.2375)\ (70 + 20) = 406$ BTU/min.
$Q_2 = 100$ BTU/min.
$Q\ = 406 + 100 = 506$ BTU/min. to be supplied.

The temperature of the incoming air would then be

$$506 = 19\ (.2375)\ (t_2 + 20)$$
$$t_2 = 92° \text{ approximately.}$$

This relatively high temperature requires that the air be diffused quite widely so that the passengers will not be subjected to discomfort.

When a liquid system is used, a pump is required to circulate the liquid. The capacity of the pump can be calculated from

$$Q = W\ C_p\ (t_2 - t_1)$$

Assume a 50° drop through the radiator. The quantity of Ethylene Glycol ("Prestone") to be circulated is then

$$506 = W\ (0.7)\ (300 - 250°)$$
$$W = 14.4 \text{ lb. per minute.}$$

The weight of the liquid at 300° is 9.3 lb. per gallon. Therefore the capacity of the pump would be $\frac{14.4}{9.3} = 1.55$ gallons per minute.

Heating Surface

The heating surface required for the boiler can be calculated from the formula

$Skdt =$ total heat to be supplied $= Q$
where $S =$ heating surface, in square feet required,
$k =$ mean coefficient of heat transmission,
$dt =$ mean temperature difference between the liquid and the exhaust gas.

In the above problem, 506 BTU were to be supplied per minute. The mean coefficient of heat transmission may be obtained experimentally. For a first approximation some value between 10 and 15 BTU per square foot per hour may be assumed.

$$dt = \frac{t_o - t_f}{\log_e \frac{t_o}{t_f}}$$

where $t_o =$ the initial temperature difference between the liquid and the gases.
$t_f =$ the final temperature difference between the liquid and the gas.

Assuming the initial temperature differences to be 1000°F., and the final temperature difference between the liquid and the gases to be 850°F.

$$dt = \frac{1000 - 850}{\log_e \frac{1000}{850}} \qquad \frac{150}{.162} = 925°\text{F.}$$

$$\text{Therefore } S = \frac{Q}{kdt} \quad \frac{506}{\frac{12}{60} \times 925} = 2.73 \text{ sq. ft., say } 3.00 \text{ sq. ft.}$$

If the exhaust pipe used for heating the liquid were 4 inches in diameter the heater would have to be $\frac{3 \times 144}{4\,\pi}$ inches or about 35 inches long.

In a similar manner, the heat transfer surface of the radiator may be calculated. The liquid temperatures and the air temperatures would be used in the latter case in the above formulas, and calculations continued as before.

Description of Typical Heating and Ventilating Systems

Schematic drawings are presented here of two essentially different heating and ventilating systems—one using air heated directly by the exhaust gases, the most available source, and the other using air heated by an intermediate or liquid heat source which has originally been heated by the exhaust gases.

In heating some of the air directly by the exhaust gases, the temperature attained is likely to be very high and it is therefore necessary to mix this heated air with a larger volume of air before it is admitted into the cabin.

With the liquid type of heater, such high temperatures are not likely to be attained so that almost all the air to be brought into the cabin is passed through the radiator and ordinarily very little cold air is required to reduce the heated air to the right temperature.

It may be worth while here to describe the four possible arrangements in detail.

Figure 29 is the diagram of a hot air heating and ventilating system without any recirculation—that is, the air is exhausted into the atmosphere without being passed through the cabin again.

A smaller entrance duct, preferably located in the leading edge of the engine cowling, or at least ahead of the engine cylinders, is conducted through the engine exhaust pipe and then out into a larger air duct designated as the main entrance duct. The small heater duct is passed through the exhaust pipe since it has been found that there is less likelihood of carbon monoxide poisoning than when a large duct encloses the exhaust pipe.

The main entrance duct may be located also in the leading edge nose cowl, or in the leading edge of the wing provided it is outside the region of possible contamination by oil fumes or exhaust gases.

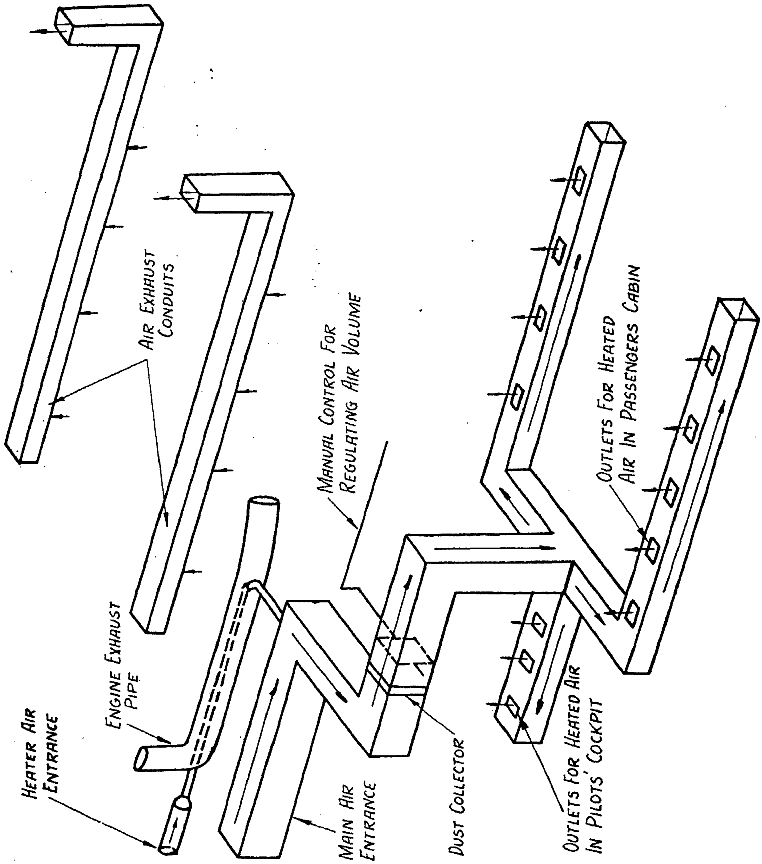


FIGURE 29. SCHEMATIC SKETCH OF A HOT AIR HEATING AND VENTILATING SYSTEM WITH NO RECIRCULATION OF AIR

A dust collector is provided to remove particles of dust or gravel that may have been kicked up by the propeller slipstream. Either in front or in rear of this dust collector there may be located a manual control for regulating the volume of incoming air if it should happen to be too much or in case it is desired to have warmer air for a short period of time. It would be desirable also to provide a regulator or shut-off in the air heater entrance duct so that no heated air would be admitted when the outside air happened to be warm enough for comfort.

The total air, heated to the proper temperature, is separated into several ducts with outlets either along the bottom or the side of the cabin. The small arrows leaving the small rectangular openings indicate the flow of air.

Exhaust ducts or conduits are placed along the top of the cabin. Instead of one central opening, it is usually desirable to have several exhaust openings to reduce the "rush" of air and its attendant noise.

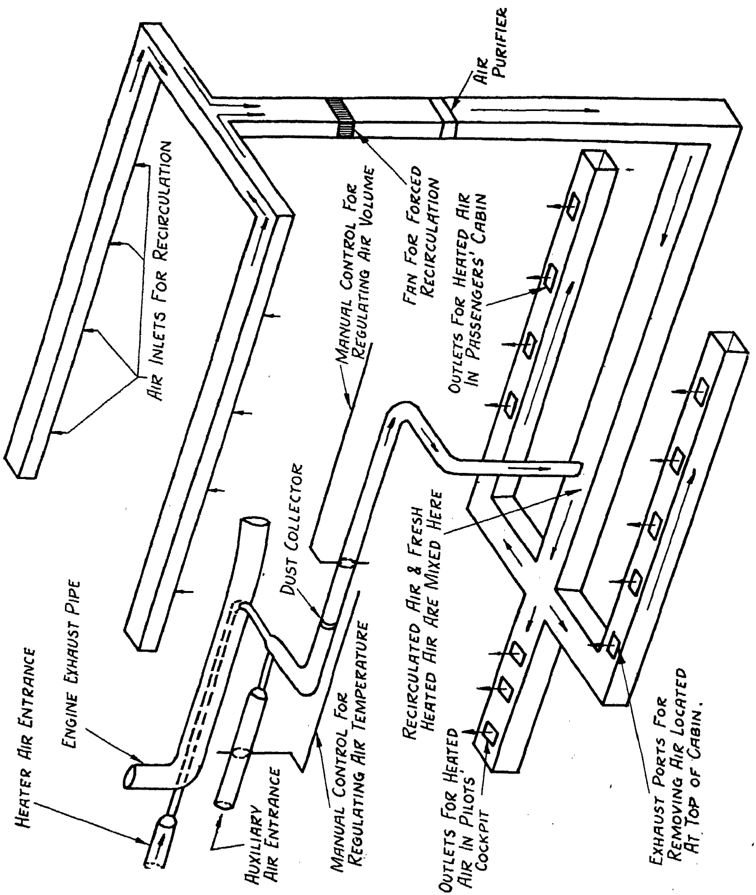


FIGURE 30. SCHEMATIC SKETCH OF A HOT AIR HEATING AND VENTILATING SYSTEM WITH RECIRCULATION OF AIR

Figure 30 is the same system designed for recirculation. The auxiliary air entrance is considerably smaller than the main air entrance, as shown in Figure 29, which it displaces since less air is needed due to recirculation. The primary purpose of this auxiliary air entrance is to regulate the temperature of the heater.

The air is circulated by means of air inlets located in the top of the cabin. The air is sucked into this duct by means of the recirculation fan which forces the air through a purifier for removing odors and then into the main duct. The entering air is at a higher

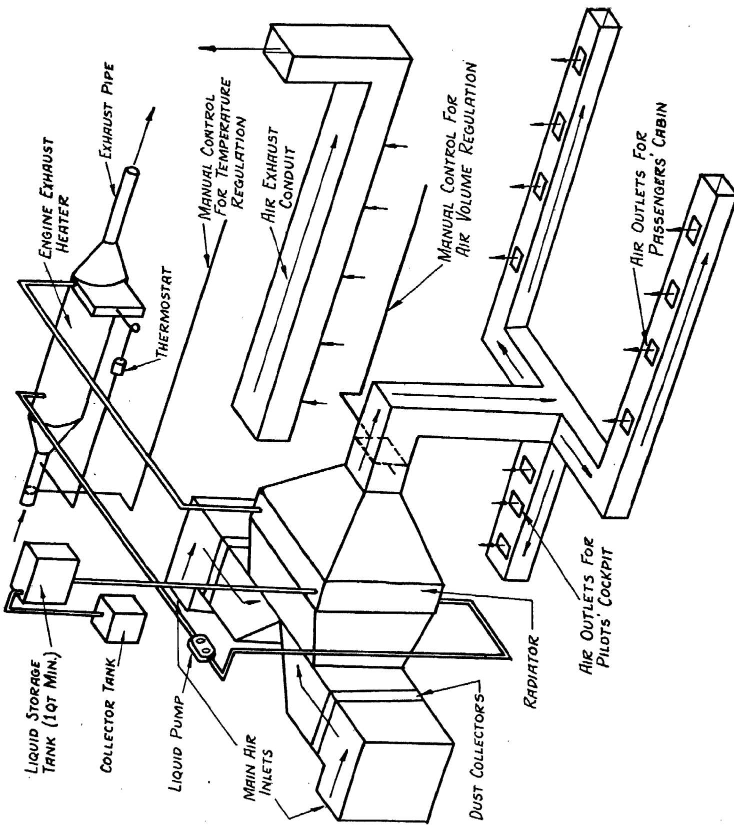


FIGURE 31. SCHEMATIC SKETCH OF A HOT LIQUID HEATING AND VENTILATING SYSTEM WITH NO RECIRCULATION OF AIR

total pressure due to the smaller size duct than the air in the recirculating duct so that the two air streams merge and are forced out of the outlets located near the bottom of the fuselage.

Exhaust ports for the air from the cabin are located in the top of the cabin. They are somewhat smaller in size than those for the non-circulating system in order to permit recirculation of the air; otherwise the air would leave the cabin as soon as it entered.

The recirculating systems do not need as large air ducts as the non-recirculating type, but these ducts are comparatively light, so that any saving in weight is more than counterbalanced by the addition of the circulating fan and a purifier unit. However, the two types—circulating and non-circulating—should be compared as to advantages for each airplane to be designed. When the airplane is very large, so that the ducts are long and the cabin to be heated and ventilated is large, then the recirculating system will probably be lighter than the non-recirculating system.

Figure 31 shows a possible arrangement for a hot liquid heating and ventilating system of the non-recirculatory type.

An engine exhaust heater or stove is built around the exhaust pipe of the engine, or if this pipe is too large, a smaller bypass exhaust pipe may be used. The latter is usually more desirable if space limitations will permit its installation since the bypass may be shut off and so put the heater out of operation when required. The diagram shows a bypass exhaust pipe.

The heater has a thermostatic control for regulating temperature as well as a manual temperature control.

The liquid to be used is in a liquid storage tank from which it flows into a radiator. The liquid in the radiator is pumped into the heater and from the heater into the radiator by means of a liquid pump driven directly or by means of a flexible shaft from a connection on the engine provided for the purpose.

A collector tank is usually connected to the storage tank since loss by evaporation would occur otherwise.

The incoming air, brought by ducts located at convenient points, is forced through the radiator and so heated to the required temperature.

A manually-operated control is located aft of the radiator to regulate the air volume.

The ventilating system is similar to the hot air system described for Figure 29.

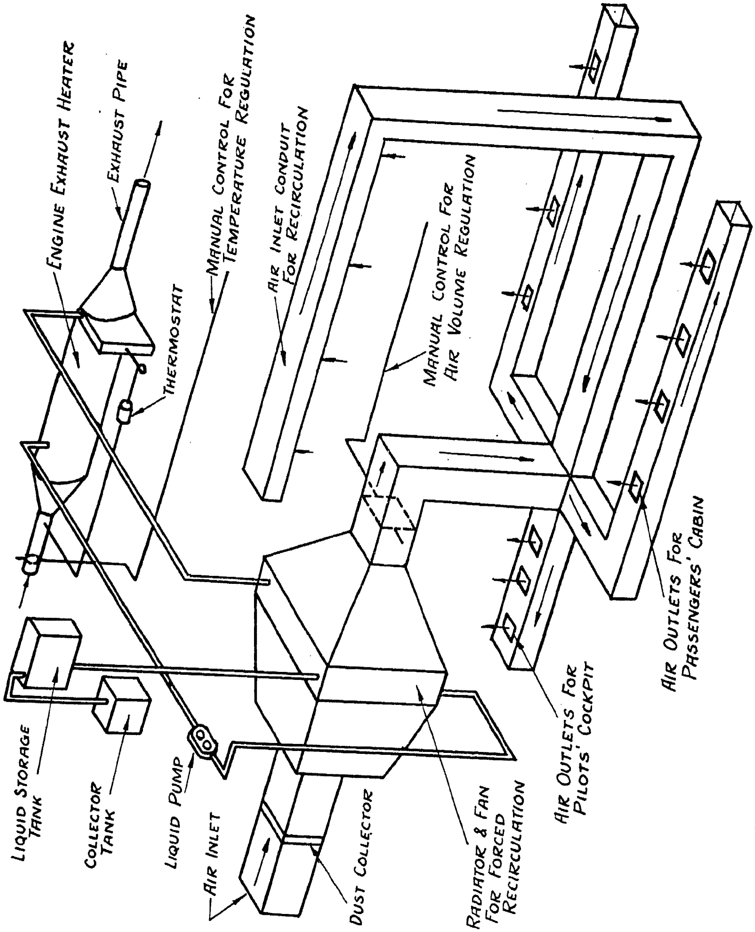


FIGURE 32. SCHEMATIC SKETCH OF A HOT LIQUID HEATING AND VENTILATING SYSTEM WITH RECIRCULATION OF AIR

Figure 32 shows a possible solution of the hot liquid recirculating system. It is similar to the hot air recirculating system, and the discussion given there holds here as well.

IX. Soundproofing

The increasing demand for more passenger comfort has brought soundproofing very much to the fore, and no airplane passenger cabin has been correctly designed unless every effort has been made to reduce noise to the practical minimum.

The problem of soundproofing an auditorium is entirely different from that of soundproofing an airplane cabin because in the former the source of noise is within so that the noise has to be absorbed. Materials must be chosen in that case which will not only absorb the noise but also avoid reverberation.

In an airplane cabin the greatest portion of the noise comes from without so that the transmission of sound must be overcome. A very thick wall is an excellent soundproofing medium but weight is important in an airplane so that materials must be chosen which will weigh the least for the greatest amount of noise reduction.

Measure of Noise

The measure of noise level is the decibel and in equation form is expressed as

$$N = 10 \log_{10} \frac{\text{Energy Input in Watts}}{\text{Energy Output in Watts}}$$

The acoustical energy is transformed into electrical energy by some suitable device—generally the "noise meter" or the "decibel meter." Since the decibel is a logarithmic relationship, it can easily be shown that if an airplane engine makes a noise equal to 100 decibels, two such engines would cause a noise level of only 103 decibels. Therefore to reduce the sound level from 100 decibels to 70 decibels, for example, requires very careful soundproofing.

Sources of Noise

The primary sources of noise encountered in the airplane are:

1. Engine exhaust.
2. Propeller.
3. Moving mechanical parts, such as gears.
4. Structural vibration.
5. Wall vibration or "drumming."
6. Whistling sounds caused by interferences.
7. Clatter due to loose structure, machinery, etc.
8. Conversation.

The noises which soundproofing attempts to alleviate are primarily engine exhaust noises, propeller noises and wall vibration. The first two mentioned cause by far the greatest disturbances.

Dr. A. H. Davis, in the *Journal of the Royal Aeronautical Society,* compares the noise of geared and ungeared engines as determined in a British airplane.

TABLE 6

Comparison of Noise of Geared and Ungeared Engines

	Geared	*Ungeared*
Propeller Tip Speed (ft. per sec.)	830	685
Propeller Diameter (feet)	9.75	10.75
Pitch (feet)	5.7	8.7
Airspeed (m.p.h.)	90-100	90-95
Average Noise Level above Threshold in Plane of Propellers (decibel)	107	91
Average Noise Level above Threshold in Rear of Cabin	95	85

These tests indicate that the reducing of the propeller tip speed helps appreciably in noise reduction. A reduction of 100 feet per second in tip speed is supposed to give as much as 10 decibel reduction in noise level.

It is not always practical to reduce the propeller tip speed unless it is already in the critical region (very close to the velocity of sound) when a reduction in propeller tip speed would also increase the propeller efficiency as well as reduce noise.

Mufflers help in reducing exhaust noises, but the level is usually not sufficiently reduced because so many other noises are present. Moreover, mufflers seriously reduce the engine horsepower and are not economical from that view point.

Effect of Frequency on Soundproofing

Experimentation with various materials reveals that all are not equally effective in reducing the incoming noise. Thus it is necessary first to analyze the frequency of the incoming noises.

Dr. Carl A. Frische in an article in the December, 1935, issue of *Aero Digest* analyzes the probable frequency distribution of noise in the pilot's cockpit of a large airplane. Table 7 has been obtained from one of his graphical illustrations.

TABLE 7

Causes of Noise	*Frequency Cycles per Second*
Propeller Unbalance Range	20-25
Engine Unbalance Range	30-40
Propeller Beating Against Fuselage	60-80
Exhaust and Engine Explosions Vibration Transmitted through Structure	200-270

After the range of frequency has been determined, sufficient materials may be chosen to form a composite blanket or wall to reduce incoming noises materially.

Table 8 below shows the effect of frequency on the decibel drop for different materials.

**TABLE 8—*Soundproofing Qualities*

Specimen	*Approx. Thickness (in.)*	*Wt. per Sq. Ft. (oz.)*	*Decibel Drop for Frequency (Noise Level 90 Decibels)*				*Wt.-Decibel Ratio Oz. per Decibel for Frequency*			
			250	500	1000	2000	250	500	1000	2000
Plywood—Felt 1	⅝	13.9	15.5	20.	19.	27.5	.897	.695	.731	.506
Plywood—Sample X	½	14.9	16.5	24.	20.	24.5	.904	.621	.745	.609
Plywood	¼	12.3	14.5	19.	18.	21.	.849	.648	.684	.586
Felt C	½	12.9	12.	13.	16.	18.	1.075	.993	.806	.717
Balsa Wood	½	7.3	12.	15.	15.	19.	.608	.487	.487	.384
Felt A	½	7.3	1.	2.	7.	6.	7.3	3.65	1.042	1.217
Felt C	¼	6.4	7.	6.	10.	10.	.914	1.067	.640	.640
Felt J	½	5.1	6.	4.	9.	11.	.850	1.275	.567	.464
Felt A	¼	3.9	2.	4.	11.	9.	1.95	.975	.355	.433
Felt J	¼	2.9	2.	1.	5.	5.	1.45	2.9	.580	.580
Sample X	¼	2.5	7.	7.	12.	21.	.357	.357	.208	.119
Sample Y	9/32(3)*	2.3	4.	4.	11.	16.	.575	.575	.209	.144
Felt 1	⅛	1.5	2.	3.	6.	6.	.75	.500	.250	.250
Felt H	⅛	1.3	.5	1.	2.5	2.	2.6	1.3	.52	.650
Sample Z	⅛	1.	2.	1.	5.	12.	.5	1.	.2	.0834
Sample Y	3/32	.75	2.	1.	5.	5.	.375	.75	.15	.15
Felt 1	⅛ (1)*	1.5	2.	3.	6.	6.	.750	.500	.250	.250
Felt 1	¼ (2)*	3.	3.5	4.5	10.	(4.)	.857	.667	.300	(.750)
Felt 1	⅜ (3)*	4.5	6.	6.5	12.5	(6.5)	.750	.692	.360	(.692)
Felt 1	½ (4)*	6.	8.5	9.5	13.5	(10.5)	.706	.632	.445	(.571)

* Number of layers. ** Data courtesy The Felters Co.

Samples X, Y and Z were kapok fibers pressed into a thin cardboard. The felts may be identified by referring to Table 9.

TABLE 9—*Grades of Felts*

Sample	*Commercial Name*	*Approx. Lbs. Sq. Yd. ¼" Thick*	*Approx. % Wool Content*
A	Cotton and Wool Pad	$2\frac{1}{8}$	85
B	All-wool Pad	$2\frac{1}{8}$	95-100
C	Wool Backcheck	$3\frac{3}{4}$	95-100
D	All-wool Pad	$2\frac{1}{8}$	100
E	All-wool Backcheck	$4\frac{1}{4}$	100
F	Soft Laundry	3	100
G	Laundry	$4\frac{1}{8}$	100
H	Grey Packing	$1\frac{1}{2}$	35
I	White Athletic Pad	$1\frac{1}{4}$	25
J	Grey Packing	$1\frac{3}{4}$	35

The soundproofing qualities given in Tables 8 and 9 were obtained from tests conducted by the author.

Soundproofing Materials

A variety of materials is used for soundproofing. Choice is determined by:

1. Location
2. Strength
3. Durability
4. Vermin-proof
5. Availability
6. Compactness
7. Adaptability
8. Cost
9. Other qualities such as heat transmission, fire-resistance, hygroscopy, etc.

The following materials have been used for soundproofing purposes:

- "Dry Zero" airplane blanket
- "Seapak"
- Balsam wool
- Felt
- Asbestos
- Bakelite
- Leather
- Balsa wood
- Rubberized hair
- Compressed cork
- Linoleum
- Micarta
- Wood
- Metal
- Doped fabric
- Upholstering material
- Plywood
- Kapok fibers

The fibrous materials, in general, seem to be the best from the point of view of noise-reduction/weight ratio. It is not desirable to use metal because of possible "drumming" and reverberation.

Application of Materials

The following are like applications for soundproofing materials:

1. *Cabin Walls*

Alternate layers of fibrous material with air spaces approximately three times the thickness of the material. One layer of the material should be attached to the outer metal skin. The inner layer may be faced with any suitable upholstery material.

Figure 33 is a schematic sketch showing the alternate layers of the fibrous material and air spaces as used for all walls and ceilings which are not to be subjected to loads and are to serve for appearance only.

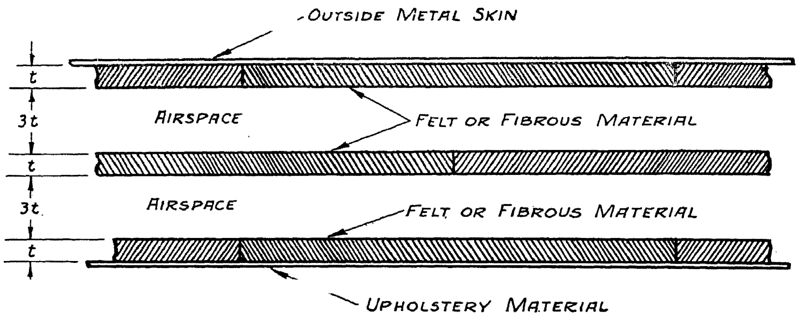


FIGURE 33. SUGGESTED TREATMENT OF CABIN WALLS FOR SOUNDPROOFING

Any fibrous material which is not very good in tension, or is likely to pull apart due to its own weight when hung, may be reinforced by muslin covers, preferably glued on in order to eliminate stitching-holes as much as possible.

Since all materials come in standard sizes, either in 36- or 72-inch widths, wall seams in the three alternate layers should not be directly behind one another.

Wherever structural members are located, they may be treated as shown in Figure 34.

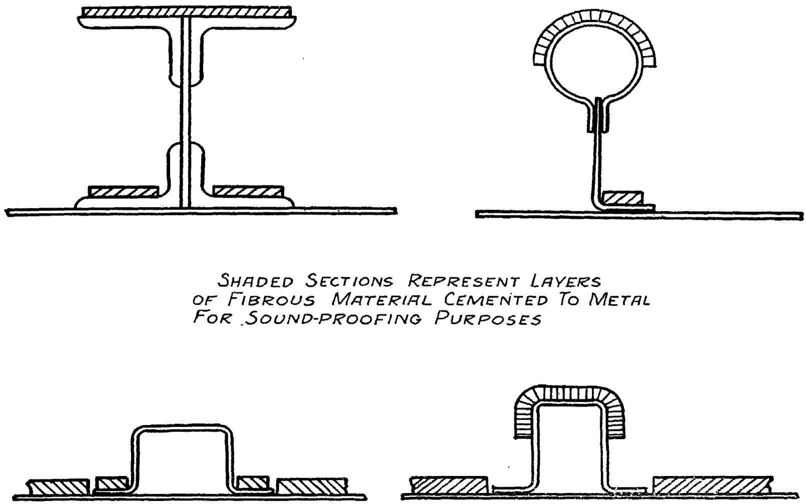


FIGURE 34. SUGGESTED TREATMENT OF STRUCTURAL MEMBERS FOR SOUNDPROOFING

2. *Windows*

One sheet instead of double panes of window glass is sufficient since the windows are far better sound-insulators than any sound-proofed portion of the cabin. The glass should be set in felt-cornered window stripping and should be supported firmly over the entire perimeter.

3. *Flooring*

A fibrous material placed over the metal floor and then covered with micarta or "battleship" linoleum forms a very effective flooring. Micarta covered panels may be made removable, for inspection of control cables and structure, by means of cowl fastener attachments.

4. *Doors, Removable Bulkheads, Baggage Compartments*

Doors and removable bulkheads may be covered with felt or some fibrous material and then protected against ordinary wear by a doped and painted aircraft linen cover. Large panels of metal in other parts of the airplane which are subjected to drumming may

be treated with small squares of fibrous materials cemented on. These are usually located in the center of the panels.

5. *Control Cables*

Control cables should be carried through a duct, preferably felt-lined. Inspection may be made by inserting suitable micarta panels at intervals.

6. *Ventilating System*

Since the ventilating ducts are exposed to the open atmosphere, special attention must be paid to them. Where the fire hazard is negligible, the ducts may be made of doped fabric. The entire lengths of the ducts, whether of metal or fabric construction, should be lined with about $\frac{1}{8}$-inch layer of felt held in place by a light-weight, coarse mesh aluminum screen.

7. *General*

Additional precautions to take in keeping down the noise level are:

a. Placement of the exhaust stack exit below the wing, for example, in a low wing monoplane, so that the wing may act as a sound "shadow."

b. Making the gap between the propeller tip and the fuselage structure as large as possible to avoid any possible drumming effect.

c. Avoiding all unobstructed holes, no matter how small—even stitching-holes are undesirable.

8. *Weights*

Weights for various materials may be found in the chapter on preliminary weight estimate (II).

Table 10 is of interest as to the allocation of various materials used for soundproofing.

TABLE 10

Total Soundproofing Weight Allowance	100%
Fibrous Soundproofing Materials	54%
Interior Fabric and Dope	11%
Sound Deadening Paint	2%
Cement for Attaching Soundproofing Material	13%
Special Reinforcements along Walls at Floor Level	2%
Cabin Door	1%
Talon Fasteners (for inspection, removal and replacements)	6%
Moulding Strips, Screws, Stiffeners, etc.	11%

Miscellaneous

In choosing a material for soundproofing, it is sometimes desirable to consider other qualities of the material.

1. *Heat Transmission*

Fibrous materials are generally good heat insulators. Table 11 below gives some representative "values of heat transmission coefficients" (BTU per degree difference at 70°F. per inch of thickness per hour).

TABLE 11—*Heat Transmission Coefficients for Felt*

Sample	*Thickness*		
	½″	¼″	⅛″
A	0.253	0.297	0.
D	.220	.272	0.155
E	.133	.182	.1425
Dry Zero	0.24 BTU/°F/in/hr		
Balsam Wool	0.25 BTU/°F/in/hr		

2. *Tensile Tests*

Additional tests by the author for the tensile and tear strength were made in order to furnish data to designers who may wish to use felt where the tensile and tear strengths are important.

Felt samples 7 in. long, 1 in. wide were used for the tests.

TABLE 12

Sample	*Cut*	*Ultimate Load (Average) Pounds*	*Cross Sectional Area Sq. In.*	*Tensile Strength Pounds per Sq. In.*
A	Lengthwise	57	.25	228
A	Crosswise	41	.25	160
B	Lengthwise	19	.25	75
B	Crosswise	31	.25	125
B	Crosswise	50	.375	133
C	Lengthwise	133	.25	530
C	Crosswise	129	.25	515
D	Lengthwise	122	.25	488
D	Crosswise	97	.25	402
E	Lengthwise	246	.25	985
E	Crosswise	173	.25	693
F	Lengthwise	217	.25	866
F	Crosswise	127	.25	506
G	Lengthwise	242	.25	968
G	Crosswise	217	.25	868

3. *Tear Tests*

For testing, the samples were partly split at one end to give two tongues, one of which was fastened in a fixed jaw clamp, the other in a movable jaw clamp to which loads were applied until the specimen began tearing.

TABLE 13—*Tearing Tests*

Sample	*Cut*	*Tearing Loads (Average)*
A	Lengthwise	4.18 lb.
A	Crosswise	2.94
B	Lengthwise	2.63
B	Crosswise	4.50
C	Lengthwise	16.75
C	Crosswise	15.40
D	Lengthwise	4.88
D	Crosswise	4.56
E	Lengthwise	22.75
E	Crosswise	23.06
F	Lengthwise	17.13
F	Crosswise	12.39
G	Lengthwise	16.54
G	Crosswise	21.69

The resistance to tearing varies with the density and the quality of the felt.

X. The Power Plant

The power plant consists of the engine, propeller, starting system, cooling system, fuel and oil systems, cowling, engine mount and miscellaneous accessories. Each item requires a considerable amount of thought for the ultimate success of the airplane depends upon the proper selection and proper functioning of every part of the power plant.

Suppose an engine is to be selected. There are air-cooled and liquid-cooled engines, either radial or in-line, or inverted in-line, or Vee, as well as a few others. If the horsepower required is rather large, the elimination of a few types may be possible since there may be no engines having a particular cylinder arrangement developing the required horsepower. Again, the air-cooled type may be preferred to a liquid-cooled type due to less complications in the installation since no cooling system is required, yet the liquid-cooled engine may be preferred for its lesser frontal area or greater reliability. The line of demarcation between air-cooled and liquid-cooled engines may be fine and one cannot say arbitrarily that one engine is better than the other until all the facts have been considered.

Then there are different arrangements of engines possible, and a complete book instead of this short chapter could be written on the power plant alone. The engines may be tractors or pushers; that is, they may be placed with the propeller in front of the engine, meeting the air before the engine; or the propeller may be placed behind the engine. Such arrangements are sometimes desired for compactness and, as in the case of tail-less airplanes, to obtain a center of gravity location reasonably far back relative to the fuselage length.

There are also tandem arrangements, or combinations of tractor and pusher arrangements; especially when the number of engines becomes so large that it may be desirable to concentrate the engines as much as possible in order to reduce the length and complication of fuel, oil and control lines.

The selection of the engine and the arrangement of a group of engines by no means ends the problem, for even the best engine cannot function properly unless attention has been paid to proper installation of the cooling system (whether it be the N.A.C.A. cowl

for air-cooled engines, or radiators and pipe lines for liquid-cooled engines); to the correct installation of the fuel and oil systems with special reference to the size of pipes or tubing; to the location of pumps and relief valves; and to the numerous little items that go to make up the whole.

In the preceding paragraphs a hasty survey of a few of the factors affecting the power plant selection and design has been made. The following material outlines the considerations to be taken into account in greater detail.

Engine Selection

Unless engines are specified in the original design specification, a study of available engines should be made, keeping the following considerations in mind.

1. Horsepower range
2. Supercharged or not supercharged
3. Critical altitudes of supercharged engine
4. Normal revolutions per minute
5. Propeller gearing
6. Weight per horsepower
7. Dependability
8. Durability
9. Economy of fuel consumption
10. Economy of oil consumption
11. Original cost
12. Ease of maintenance
13. Type of cylinder arrangement
14. Overall dimensions
15. Method of cooling
16. Department of Commerce approval
17. Availability

Specifications of American engines are given in Table 14.

It is comparatively easy to determine the approximate maximum horsepower required on the basis of horsepower loading and the estimated gross weight of the airplane to be designed and built.

$$\text{Required Horsepower} = \frac{\text{Gross Weight}}{\text{Horsepower Loading}}$$

Some of the considerations listed above may be difficult to ascertain, especially for the student, and may be safely disregarded. It is well to keep these various considerations in mind, however, for possible future use.

Engine Mount

1. Figures 35 A, B and C show possible engine mounts for the radial and in-line engines.

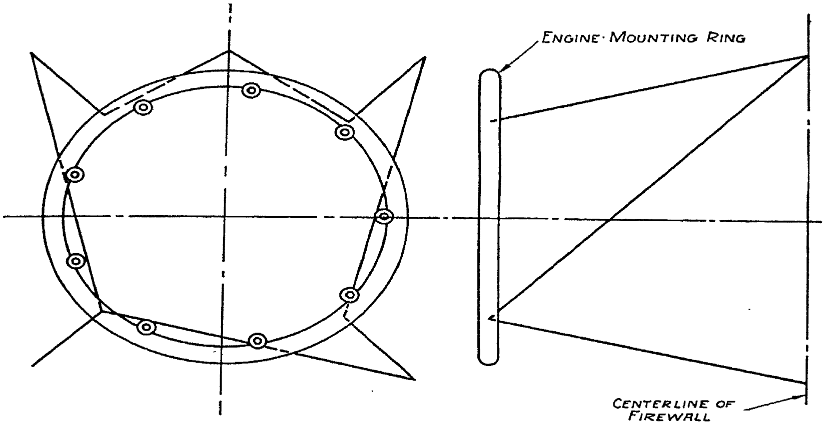


Figure 35 A. Diagrammatic Sketch f 'r an Engine Mount for a Radial Aircraft Engine

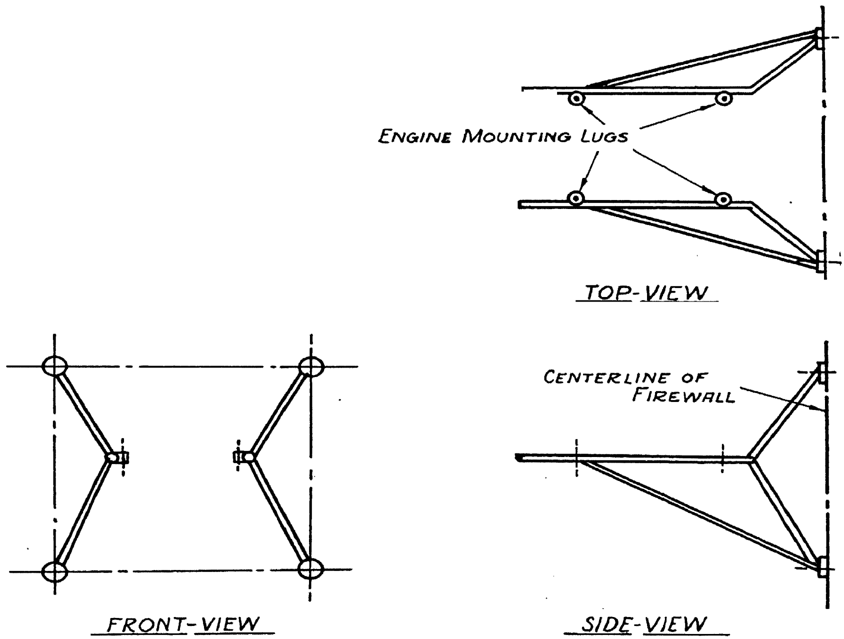


Figure 35 C. A Diagrammatic Sketch of an Engine Mount for an In-line or Vee Engine

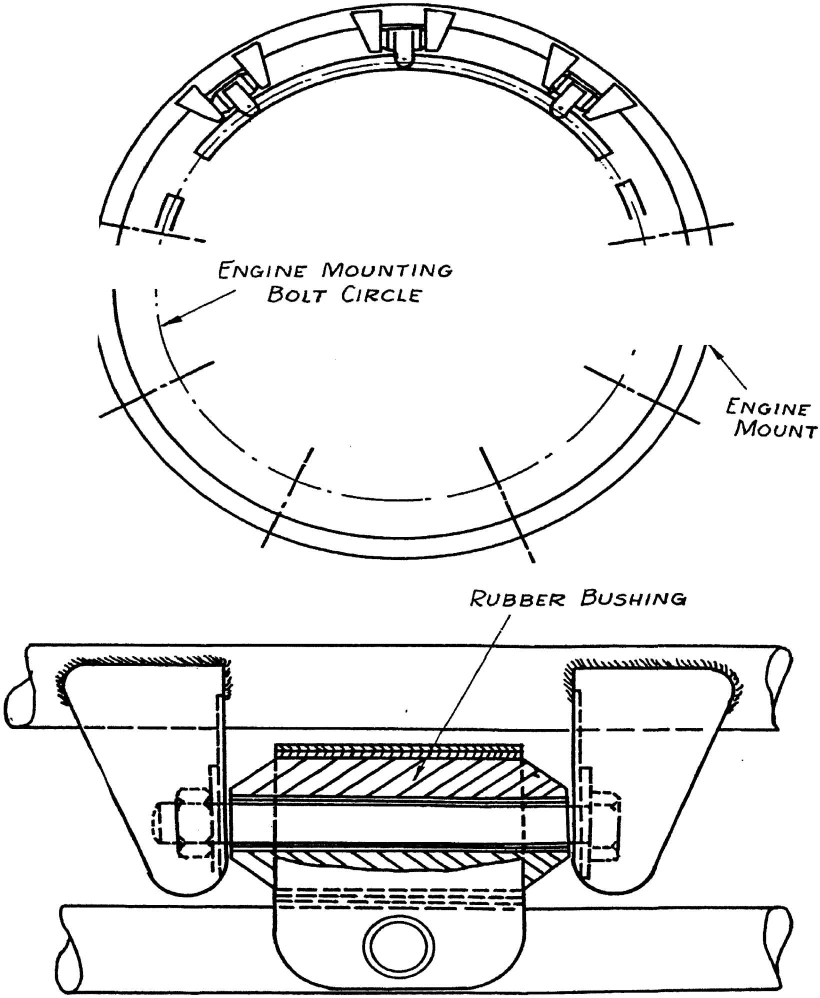


FIGURE 35 B. A MODIFICATION OF THE MOUNT SHOWN IN FIGURE 35 A. RUBBER VIBRATION MOUNTING IS INCORPORATED IN THIS DESIGN

2. More and more attention is being paid to reduction of transmission of vibration. Rubber mounting of the engine on the engine mount and of the engine mount to the fuselage are the usual means of reducing the vibration.

Figures 36 A and B show some forms of these types of mounting.

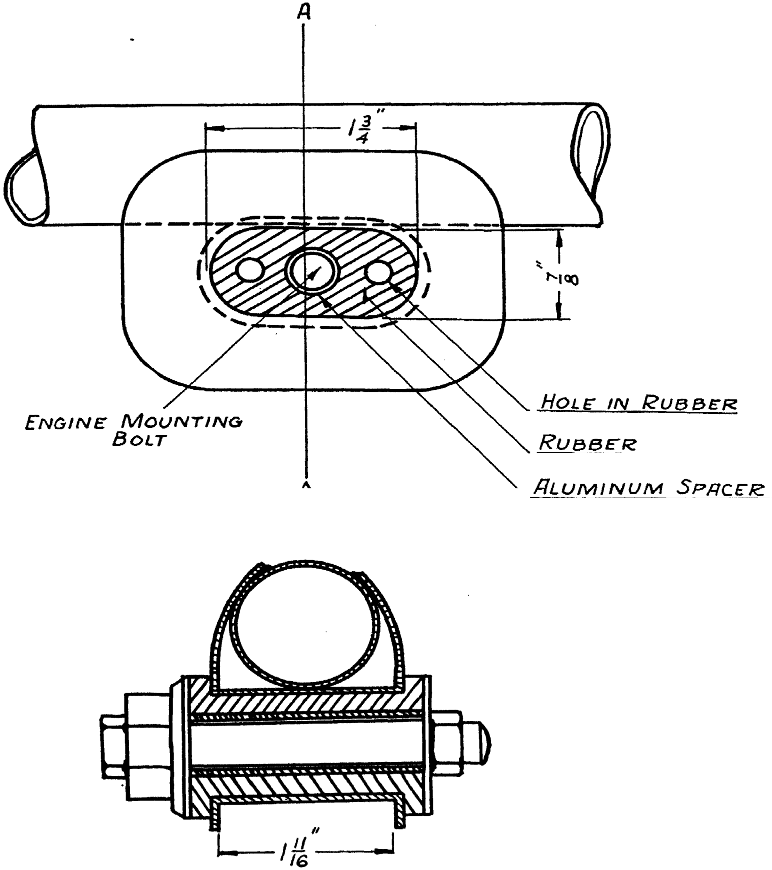


FIGURE 36 A. SUGGESTION FOR A RUBBER VIBRATION MOUNTING AT THE ENGINE MOUNTING RING

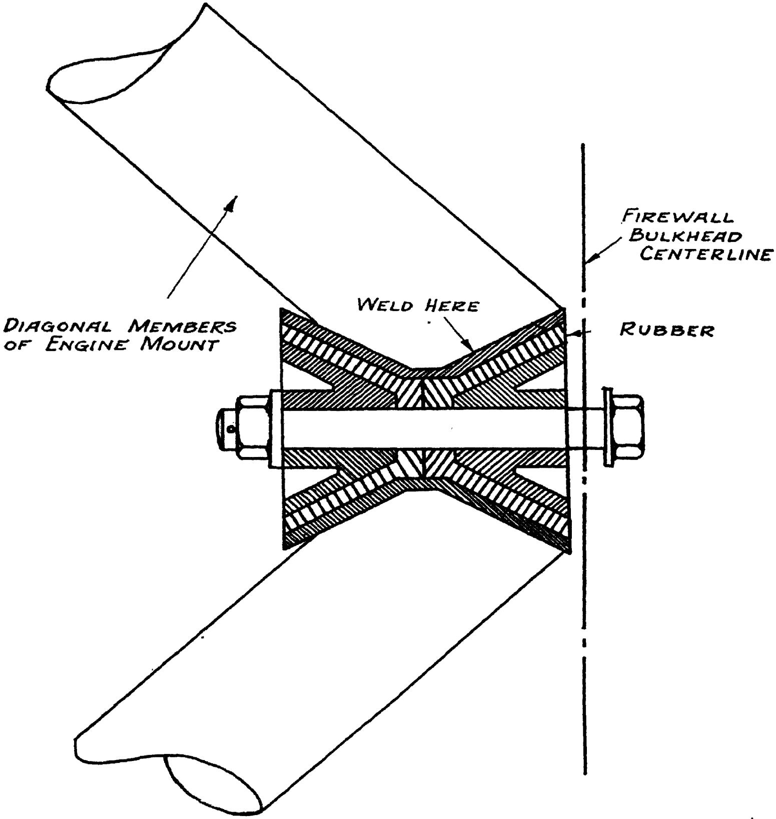


FIGURE 36 B. A POSSIBLE RUBBER VIBRATION MOUNTING AT THE FIREWALL FOR THE ENGINE MOUNT

The design should be modified so that the bolt attaching the mount to the firewall bulkhead is not in tension

3. Tubular steel is customarily used for the entire mount whether the airplane is to be reinforced monocoque or not. The sizes of the members are determined by stress analysis, although $1\frac{1}{8}$ in. or $1\frac{1}{4}$ in. outside diameter tubing may be used for preliminary design until a stress analysis has been made.

4. Care should be taken that all members leading from the engine mounting ring to the firewall clear all accessories and permit easy access to all necessary parts.

5. The engine mount should be so constructed that the entire power plant, including the oil system cowling and manifold, may be readily detached for replacement.

Firewall

1. The firewall is located as close to the engine as the removal of the rearmost accessory located on the engine will permit.

2. No firewall need be provided when the engine is located in a separate nacelle unless it is intended to locate fuel in the same nacelle.

3. The firewall should isolate the engine compartment completely. All openings required for cables, rods and the like should be fitted with close fitting grommets or bushings. Control systems passing through the firewall should be so designed that their motion is axial or rotational in order to avoid large openings.

4. Any adjacent inflammable structural members should be protected by asbestos or some equivalent insulating material.

5. Firewalls are made either of ferrous metal of suitable thickness, or aluminum (or aluminum alloy) used for outside sheets of an asbestos sandwich. The following materials for the firewall are recommended.

a. A single sheet of terne plate (70% tin plus 30% lead) not less than .028 inch thick.
b. A single sheet of stainless steel not less than 0.015 inch thick.
c. A single sheet of aluminum alloy either 0.032 or 0.040 inch thick.
d. Two sheets of aluminum or aluminum alloy not less than 0.02 inch thick fastened together with an asbestos paper or fabric sheet at least $\frac{1}{8}$ inch thick between them.

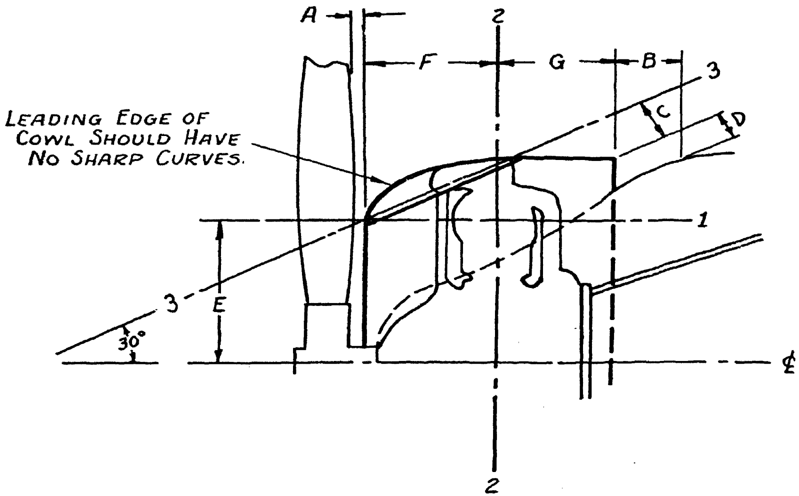
6. It is important that the fire bulkhead fit closely to the cowling all around the section of the fuselage.

Cowling

1. The type of cowl developed by the N.A.C.A. has met with general favor for radial engines by the industry. In designing the contours of any type of engine cowling, special care should be exercised to avoid all abruptness or reversal in curvature. A well-rounded contour will help effectively in reducing parasite resistance.

2. The correct proportioning of the engine cowls depends much upon the particular engine, engine nacelle, past experience, likely operating conditions, etc. Figures 37, 38 and 39 contain suggestions to the designer.

3. Engine cowling should be fastened to the mount and not to the fuselage, and in such a fashion that the major portion may be removed in a few minutes for maintenance, inspection and minor repairs.



AXIS 1–1 ~ LOCATE AS CLOSE TO COOLING FINS AS POSSIBLE.
AXIS 2–2 ~ CENTERLINE OF CYLINDERS. CHOOSE FRONT BANK OF CYLINDERS FOR A TWIN-ROW RADIAL ENGINE.
AXIS 3–3 ~ DRAWN AT 30 DEGREES TO THE ENGINE CENTER-LINE THRU INTERSECTION OF AXIS 1–1 & AXIS 2–2.

DIMENSIONS:

A ~ MINIMUM PROPELLER CLEARANCE OF ONE INCH; PREFERABLY 2 INCHES. IF PROPELLER CLEARANCE CAN NOT BE EASILY DETERMINED, USE TANGENT DRAWN TO CRANKCASE AT PROPELLER SHAFT.
B ~ WIDTH OF SLOT = [2×H.P.] ÷ CIRCUMFERENCE
C ~ MINIMUM OF 4 INCHES.
D ~ " " 3 "
E ~ SEE INSTRUCTIONS FOR AXIS 1-1 ABOVE.
F ~ FROM $\frac{2}{5}$ TO $\frac{1}{2}$ G.
G ~ FROM 4 TO 8 INCHES BEHIND CENTERLINE OF MOUNTING RING.

FIGURE 37. SUGGESTIONS FOR DEVELOPING AN N. A. C. A. COWL FOR A RADIAL ENGINE

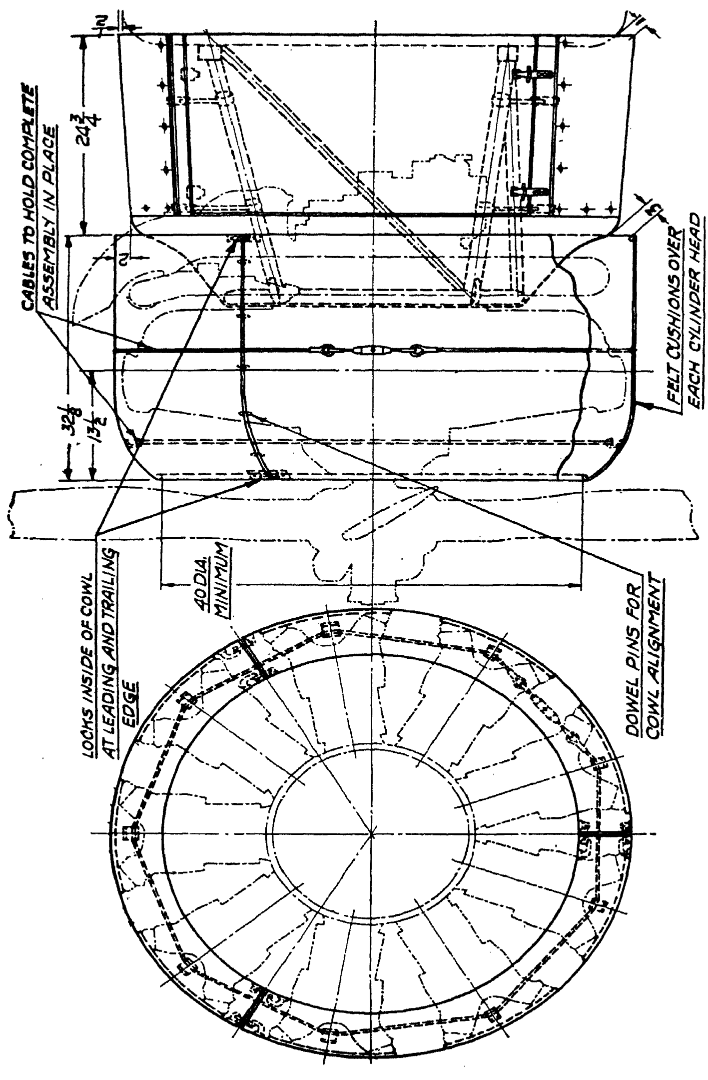


FIGURE 38. SUGGESTED COWLING FOR THE WRIGHT CYCLONE ENGINE

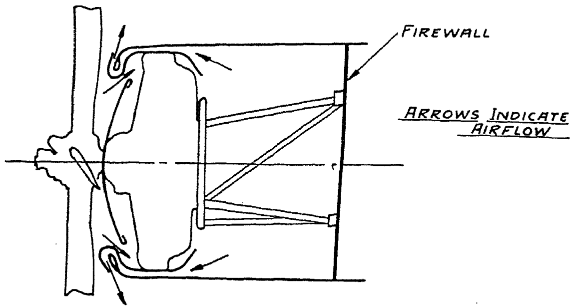


FIGURE 39. THE "REVERSED FLOW" COWL FOR RADIAL ENGINES

4. The cowling may be made of aluminum alloy—450, 2S or 52S in thickness, varying from .032 to .050 with the larger thickness usually reserved for the cowl ring proper and the smaller, or some intermediate thickness for the rest of the cowling.

5. The engine cowling should be suitably ventilated to prevent accumulation of gases.

6. All cowling around the power plant and on the engine side of the firewall should be made of metal and so designed that any accumulations of dirt, waste or fuel may be observed without complete removal of the cowling.

7. The cowling must fit tightly to the firewall, but openings may be provided if the airplane surface within 15 inches is protected with metal or other suitable material.

8. Unless small units of the cowling may be removed easily for inspection and repair purposes, it is desirable to provide properly secured small access doors in the cowling at suitable points.

9. The cowling should be completely drained in all attitudes of flight and on the ground, with separate drains provided for the parts of the fuel system liable to leakage.

10. All drains should be so located as to prevent fuel or oil from dripping onto the exhaust manifold or any parts of the aircraft, or permeating any cellular material.

11. Carburetor air intakes must open entirely outside of the cowling, unless the emergence of back-fire flames is positively prevented. The air intakes should be suitably drained.

Exhaust Manifolds

1. The exhaust manifolds, stacks or collectors preferably should be made of interchangeable sections, usually of 18-8 corrosion resisting steel—Iconel—or carbon steel. The wall thicknesses vary from 0.035 to 0.049 inch with the greater thickness reserved for high-powered engines, particularly those using high octane fuel.

2. The diameters of the manifolds vary from 2 to 4 inches, and may be bent with an inside radius as small as two diameters although a larger radius is customarily used for ease in fabrication as well as to reduce undesired back pressures. The cross-sectional area of the exhaust manifolds should gradually increase until the cross-section at the last cylinder is at least 50 per cent of the total exhaust-port area of the particular engine for which the manifold is being designed.

3. The "downwind" clearance of the open end of the exhaust pipe should be at least 4 feet. Any exhaust pipe which is not exposed to the outside air should be either water or air cooled by a special cooling system surrounding the unexposed pipe.

4. The exhaust pipe should be kept at least 3 inches away from any inflammable part of the airplane, and the exhaust end should be at least 5 inches away from any inflammable part.

5. Expansion joints should be provided for and such joints should be capable of articulation to a certain degree to permit changes not only in length but in angle of alignment also due to expansion.

6. Gases should be discharged clear of the airplane structure, and so that they will not blow back on the carburetor air intake, the pilot or passengers, nor cause a glare ahead of the pilot at night.

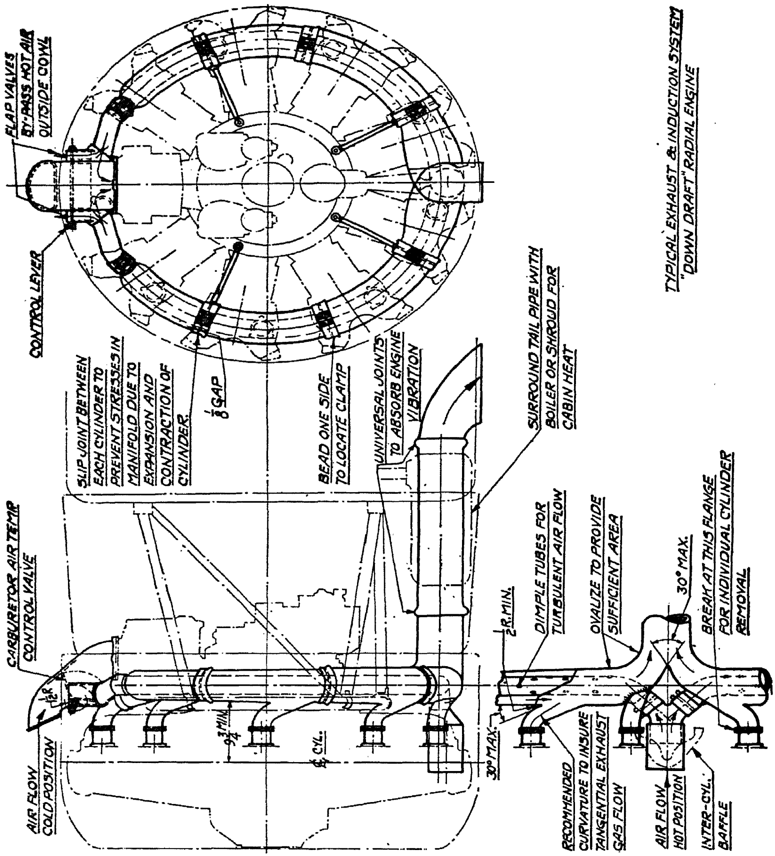
7. Figure 40 shows a typical combined exhaust and induction system for a "down-draft" radial engine.

Fuel Systems

The fuel system consists of tanks, piping exterior to the engine, pumps not integral with the engine, strainers, gauges, pipe fittings and valves and cocks.

Diagrammatic layouts of typical fuel systems are shown in Figures 41, 42 and 43.

These figures merit a brief description. The sketch in Figure 41 is a schematic representation of a fuel system employing a single



Courtesy Wright Aeronautical Corp

FIGURE 40

tank which may be located in the center section of a high wing monoplane, or on the side of a fuselage, or in the butt sections of a low wing installation.

The primer is operated from the instrument panel board in the pilot's cockpit. A shut-off cock is located in the primer line.

For starting the flow of fuel, it is necessary to use the hand or wobble pump which should be as far below the fuel tank as possible so that it is flooded at all times. As soon as the engine is started, the

engine driven gear pump will continue the pumping of fuel. The hand or wobble pump is so arranged that fuel will flow through it even when it is not being operated. If its location is at some distance from the cockpit it may be operated through a suitable linkage system to a crank near at hand to the pilot.

The engine driven gear pump has a bypass so that in case of failure, the hand or "wobble" pump becomes the emergency pump and will force the fuel past the gear pump through the bypass.

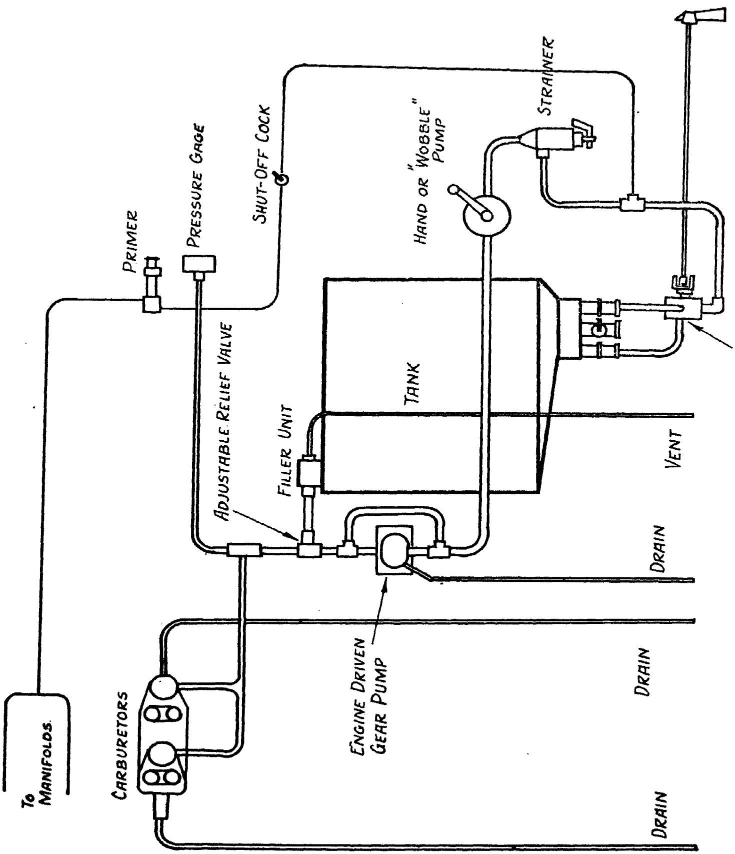


FIGURE 41. FUEL SYSTEM SHOWING ONLY ONE MAIN FUEL TANK

In case the pump supplies the fuel at a greater pressure than the carburetors are designed to take, the adjustable relief valve permits the fuel to flow back into the tank.

All tanks must be vented in order to prevent partial vacuums from forming since these may reduce or prevent proper fuel flow. This vent is usually located at the topmost point—usually the filler unit—of the gasoline tank.

A strainer must be included in any fuel line and is usually located at the lowest possible point in the line and in a place where it is easily accessible, for the strainer collects the water, grit and dirt which may have collected in the fuel system even with the best of precautions.

The three port, two-way cock is operated from the cockpit.

Figure 42 is similar to the single tank layout just discussed except for the addition of a gravity fuel tank. This system is especially popular for small sport biplanes where it is possible to put a gravity fuel tank in the center section of the cabane structure. The gravity tank is so called because it is high enough above the carburetors of the engine so that the fuel will flow without the use of a pump.

A pressure gauge is used in all fuel systems in order to indicate the pressure in the fuel lines just ahead of the carburetors at all times. If the pressure should fall below a pre-determined value, it is usually a sure sign of danger—either the line is clogged or the fuel has given out.

Since multi-engine designs are quite common, Figure 43 is of interest for it shows a schematic fuel system diagram of a twin engine installation.

Specific recommendations for the design of the fuel system are:

1. *General*
   A. Air pressure fuel systems are not to be used—either straight gravity feed or mechanical pumping of fuel only is permitted.
   B The system shall be arranged so that the entire fuel system may be utilized in the steepest climb and at the best gliding angle and so that the feed parts will not be uncovered during normal maneuvers including moderate rolling or slide slipping.

2. *Pumps*
   A. If a mechanical pump is used, an emergency hand pump is also required.
   B. Hand pumps may be used for pumping fuel from an auxiliary to a main tank.

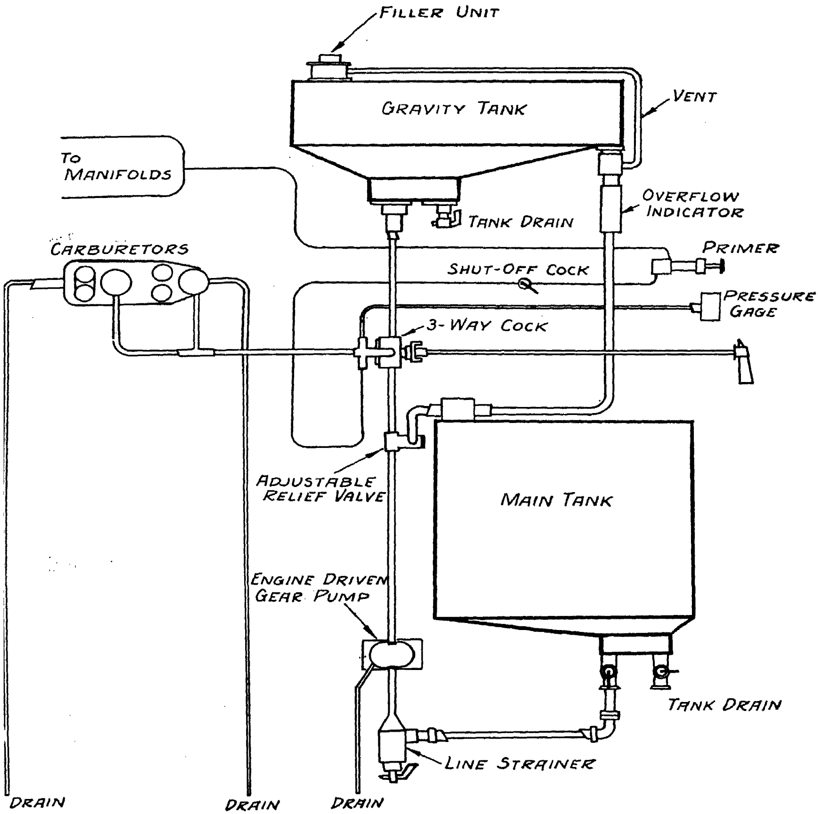


FIGURE 42. FUEL SYSTEM WITH A GRAVITY AND A MAIN FUEL TANK

C. The hand or "wobble" pump should be placed at least 50 per cent below the top of the main fuel supply. In some cases, it may be desirable to place it as far below the fuel system as the design will permit. Operation of the pumps may be done from the cockpit by means of a suitable push-pull operating system.

D. The hand-operated pump must be so installed as to be operated readily from the cockpit without requiring any opening of valves or cocks in the system.

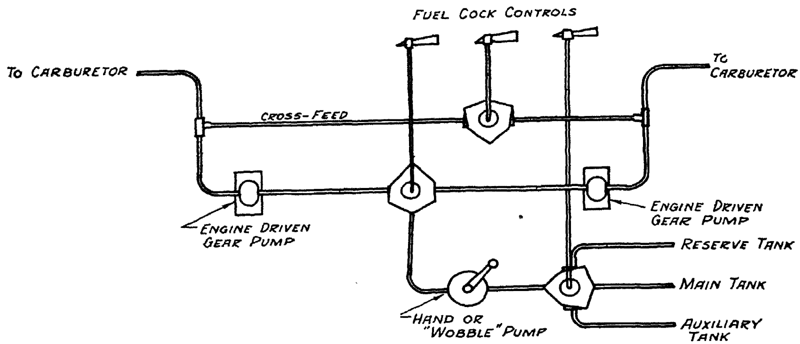


FIGURE 43. PART OF THE FUEL SYSTEM FOR A TWIN-ENGINE INSTALLATION

The fuel tanks are similar in design to those shown in Figures 41 and 42.

3. *Tanks, Vents, etc.*

A. Fuel tanks should be capable of withstanding an internal test pressure of $3\frac{1}{2}$ pounds per square inch without failure or leakage. Fuel tanks which have a maximum fuel depth greater than 2 feet should be investigated for the pressure developed during the maximum applied acceleration with full tanks.

B. Welded fuel tanks of 2S, 3S or 52S, aluminum alloy are common. Riveted tanks of 17ST aluminum alloy have also been made. Shot welded stainless steel fuel tanks have been made unusually light.
The thickness of the aluminum alloy varies from 0.040 to 0.065 inch in thickness depending upon the size and manner of support.

C. Fuel tanks with curved sides seem to be preferred to fuel tanks having large flat sides.

D. Baffle plates welded to the sides of the tank seem to be preferred to those riveted to the sides.

E. The tanks should be so installed that easy removal or inspection is possible.

F. No fuel tank may be closer to the engine than the remote side of the firewall.

G. Surfaces of the tank or tanks should be so ventilated that the fuel fumes cannot accumulate in case of leakage. At least $\frac{1}{2}$ inch air-space should be allowed between the tank and the firewall.

H. The tanks should not be supported by leather or rubber straps (or fabric containing rubber). Felt cushions are recommended. In general, the tanks should be supported so that the weight is distributed throughout the entire tank structure.

I. Each tank must be suitably vented from the top portion of the air space. Such air vents should be arranged so as to minimize the possibility of stoppage by ice formation.

J. If two or more tanks have their outlets interconnected, the air space in the tanks should also be interconnected to prevent differences in pressure at air vents of each tank of sufficient magnitude to cause fuel flow between tanks.

K. Where large fuel tanks are used the size of the vent tubes should be so proportioned as to permit rapid changes in internal air pressure and thereby prevent collapse of the tanks in a steep glide or dive.

L. Each fuel tank should be provided with a pump and a drain located at the lowest point when the airplane is in the normal position on the ground. The main fuel supply shall not be drawn from the bottom of this pump.

M. In case of aircraft employing more than one fuel supply, suitable provision shall be made for independent feeding from each supply.

N. All tank outlets should have cocks.

O. Line strainers, cocks, hand pumps, relief and bypass valves should be placed a minimum of 50 per cent below the top of the main fuel supply.

P. Filler caps preferably should be mounted on a large manhole plate which, in turn, is mounted on the tank by means of anchor nuts and gaskets. This permits less service work and aids in the cleaning of the interior of the tank when required.

Q. One or more positive and quick-acting valves that will shut off all fuel from the engine should be within easy reach of the pilot or flight mechanic.

R. When fuel tanks are equipped with dump valves, the operating mechanism for such valves should be within convenient reach of the pilot or flight mechanic and the outlets should be so located as to preclude the possibility of fire when they are operated.

S. The minimum available gasoline capacity should be at least 0.15 gallon per rated engine horsepower. If the fuel is other than gasoline, the available fuel capacity with full payload should be sufficient for a two hour flight at cruising speed.

4. *Fuel Gauges*

A. A satisfactory gauge must be installed on all aircraft indicating to the pilot or flight mechanic the fuel in each tank while in flight.

B. Installation drawings and dimensions of one type of fuel level gauge are shown in Figure 44.

C. Where two or more tanks are interconnected and it is impossible to feed from each one separately, only one fuel level gauge need be installed.

D. If a glass gauge is used, it shall be suitably protected against breakage.

5. *Strainers*

A. One or more strainers of adequate size and design incorporating a suitable sediment trap and drain should be provided in the fuel line between the tank and the carburetor and should be installed in an accessible position so that the screen may be removed easily for cleaning.

B. It is desirable to install a 100-mesh filter and sediment bulb in the fuel line immediately adjacent to the carburetor.

C. Another filter and sediment bulb should be installed between the fuel pumps and as near to the main tank as possible.

6. *Piping and Fittings*

A. All fuel piping and fittings should be of such size that under pressure of normal operation the flow is not less than double the normal flow required for full engine power. A test for compliance with this requirement may be necessary in certain cases.

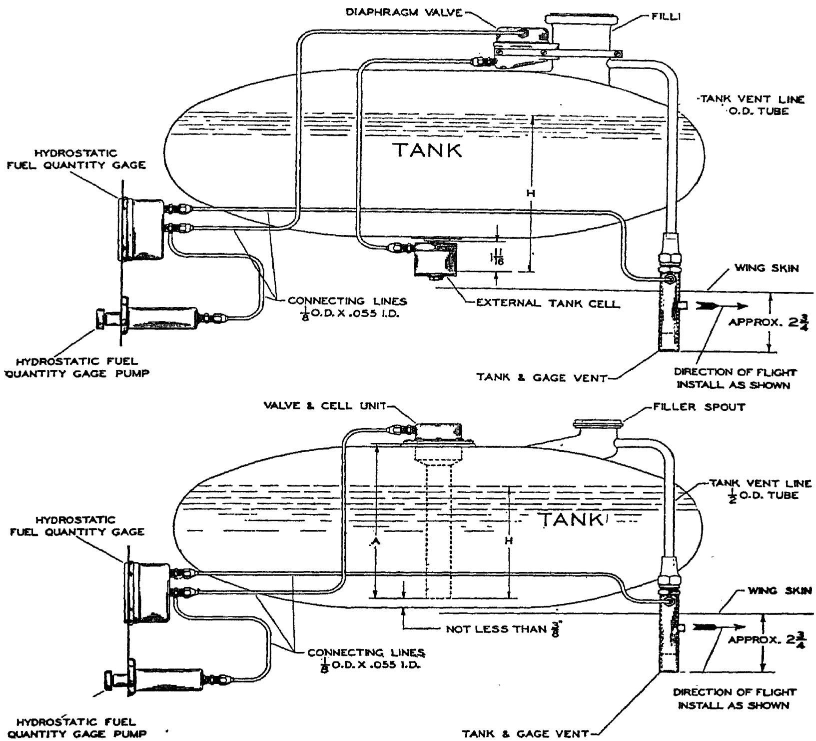


FIGURE 44. INSTALLATION OF THE KOLLSMAN FUEL QUANTITY GAUGE

B. Recommended sizes for fuel lines are listed in Table 15 below.

TABLE 15

Purpose	*Fuel to be Delivered*	*Outside Diameter*	*Wall Thickness*
1. From Tank to Fuel Pumps	Up to 60 gal. per hr. From 60 to 100 gal. per hr. From 100 to 150 gal. per hr.	½ in. ⅝ in. ¾ in.	Not less than 0.040 in.
2. (a) Pumps to carburetor, (b) carburetor to overflow drains	Up to 30 gal. per hr. 30 to 60 gal. per hr. 60 to 100 gal. per hr. 100 to 150 gal. per hr.	⅜ in. ½ in. ⅝ in. ¾ in.	Not less than 0.032 Not less than 0.040 in.
3. Fuel Pumps Relief Lines	Same as for Item 1		
4. Primer Tubing		⅛ in.	Not less than 0.032
5. Vents		¼ in.	Not less than 0.032

For feed lines from gravity tanks to carburetors, the piping should be of sufficient size to permit flow of 50 per cent in excess of the fuel required by the engine at full throttle under gravity head equal to that obtainable in the particular airplane in any flying airplane.

For tank overflow, the piping should be of sufficient size to permit flow of total pump capacity under a gravity head equal to the one existing in the airplane.

The above recommendations have been simplified by some commercial operators to the point of specifying piping $\frac{1}{2}$ inch outside diameter, by 0.035 inch wall, for engines developing less than 600 horsepower, and $\frac{3}{4}$ inch outside diameter, by 0.042 inch wall, for engines developing more than 600 horsepower.

C. Flexible tubings have larger outside diameters than those specified above, due to their manner of construction.

D. 4SO and 52SO aluminum alloy, copper or various types of flexible tubing are used for fuel lines.

E. Bends of small radii are not permitted. A radius of at least 3 diameters is recommended.

F. Copper fuel lines which have been bent before installation should be annealed.

G. Vertical bumps in the line should be avoided.

H. A flexible connection or its equivalent is required between the part of the fuel system attached to the engine and that attached to the primary structure of the airplane.

I. Flexible hose connections must have metal liners.

J. Flexible fuel lines should be suitably supported. The spacing of such supports depends on the type of fuel line used, and, in general, varies between 6 and 12 inches.

K. Except for the flexible connections specified, union connections may be used for all other lines.

L. Fuel lines should be easily accessible for inspection and should be readily removable without disturbing the primary structure.

Figure 45 shows typical oil and fuel line connections.

Lubricating System

The lubricating system consists of the oil tank or tanks, oil pumps not integral with the engine, temperature regulators, piping, fittings, valves, shut-off cocks and strainers.

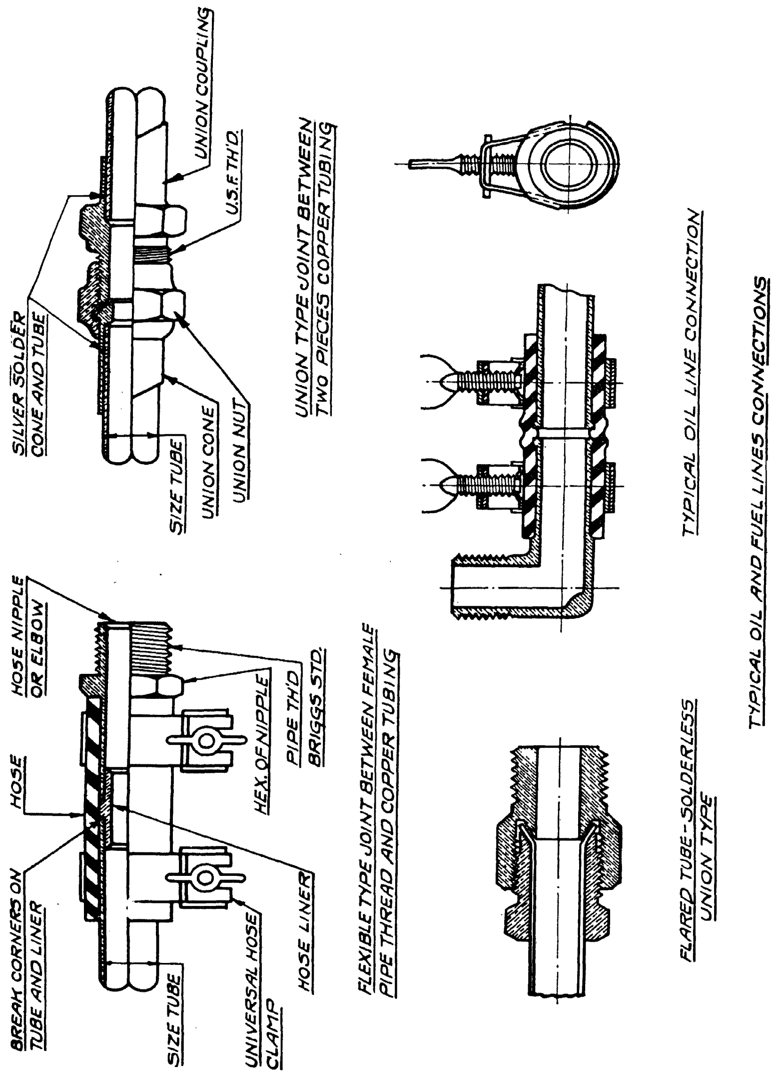

SILVER SOLDER CONE AND TUBE
UNION COUPLING
U.S.F. TH'D.
SIZE TUBE
UNION CONE
UNION NUT
UNION TYPE JOINT BETWEEN TWO PIECES COPPER TUBING
BREAK CORNERS ON TUBE AND LINER
HOSE
HOSE NIPPLE OR ELBOW
SIZE TUBE
UNIVERSAL HOSE CLAMP
HOSE LINER
HEX. OF NIPPLE
PIPE TH'D BRIGGS STD.
FLEXIBLE TYPE JOINT BETWEEN FEMALE PIPE THREAD AND COPPER TUBING
TYPICAL OIL LINE CONNECTION
FLARED TUBE - SOLDERLESS UNION TYPE
TYPICAL OIL AND FUEL LINES CONNECTIONS

FIGURE 45

Figures 46, 47 and 48 show typical oil systems for air-cooled engines.

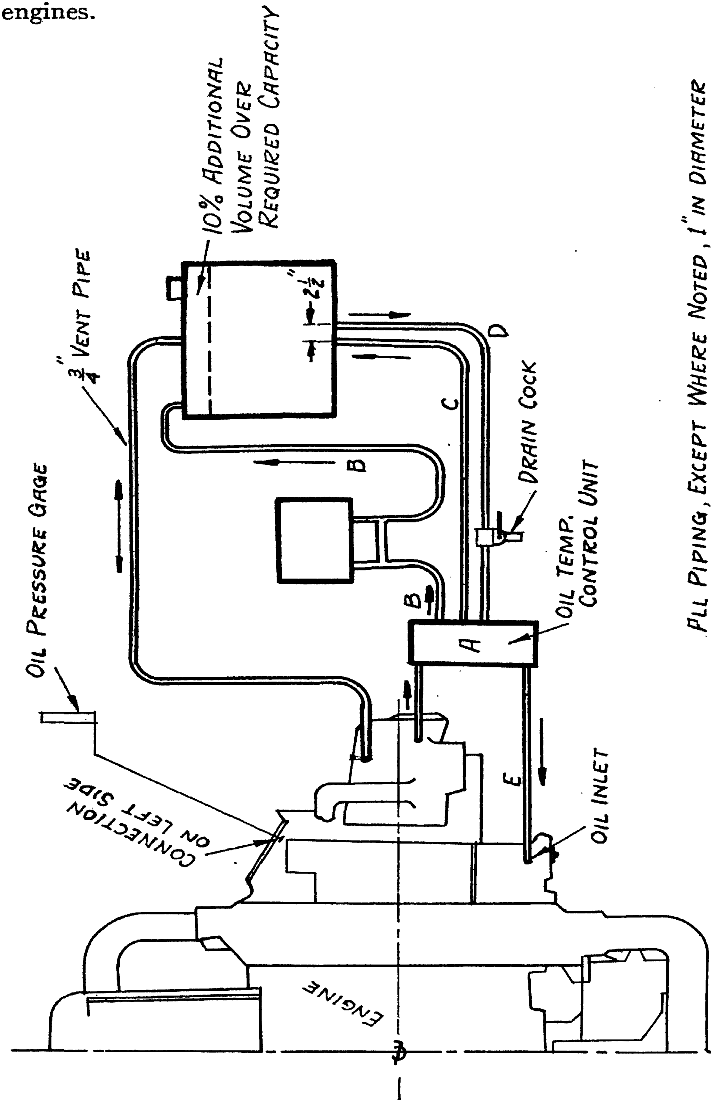
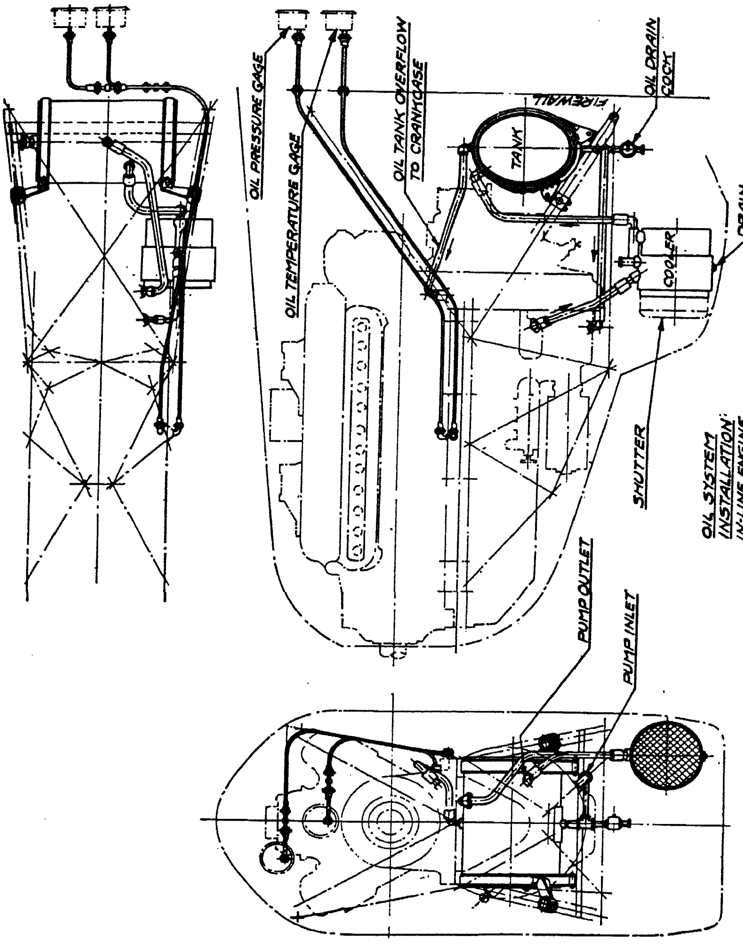


FIGURE 46 A. AN OIL SYSTEM FOR A WHITNEY RADIAL ENGINE



Suggested by Wright Aeronautical Corp.

FIGURE 46 B

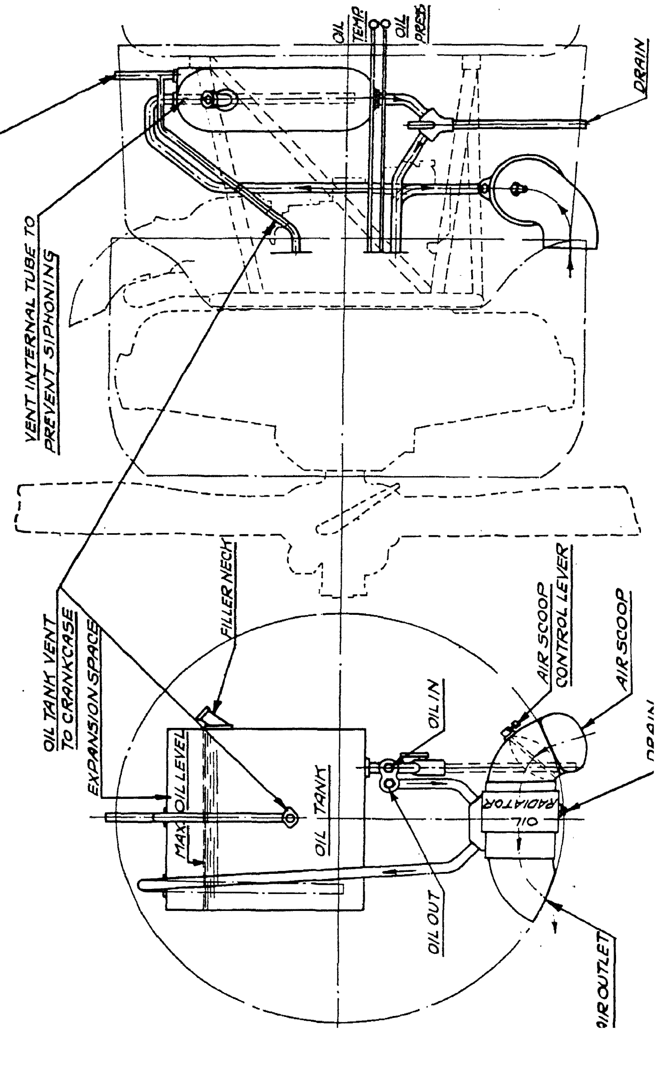


FIGURE 46 C. TYPICAL OIL SYSTEM FOR WRIGHT AERONAUTICAL CORPORATION AIR COOLED ENGINES

1. *Tanks*
   - A. The oil capacity of the system should be at least 1 gallon for every 16 gallons of fuel but should not be less than the minimum specified for safe operation of engine. Commercial operators use the formula: oil required equals 10 gallons plus 1 gallon of oil for every 20 gallons of fuel.
   - B. The oil tank should allow for at least 10 per cent volume over that required for the oil alone to provide for expansion space.
   - C. Oil tanks must be capable of withstanding an internal test pressure of 5 pounds per square inch without failure or leakage.
   - D. Oil tanks are made of the same materials as those used for the fuel tanks—2S, 3S, or 52S aluminum alloy with wall thicknesses varying from 0.040 to 0.065 inch.
   - E. The lubricating system should be so arranged that delivery of oil to all the necessary parts of the engine is maintained with the engine running at any speed from idling to full power, and in any reasonable altitude of the airplane whether in flight or at rest.
   - F. The oil tank should be located as near to the oil pump as possible, and generally on the same side of the firewall as the engine.
   - G. The bottom of the oil tank should be above the level of the oil pump.
   - H. A suitable means should be provided to determine the amount of oil in the system.
   - I. All oil tanks should be suitably vented.
   - J. The oil temperature should be measured at the engine inlet.
   - K. Where possible, all tank pipe connections should be grouped on a removable plate mounted on suitable gaskets.
   - L. The filling pipe should have 2 inches inside diameter.
   - M. The filler opening should be so located as to permit filling from the outside of the fuselage or engine cowling without the use of a special type of funnel.
   - N. Care should be taken in the design to prevent any overflowing oil from falling upon wiring, engine or parts of the structure that may be damaged.
   - O. Oil tanks should be so supported as to distribute the weight through the entire structure.

P. Leather or rubber should not be used to support the oil tank in its cradle.
Q. Provision should be made for readily removing the tank or tanks without disturbing either the engine or the primary structure.
R. Oil tanks should be vented to the engine crank case or overboard if provision is made to prevent air pockets from forming in the lines or in the tank.
S. Means should be provided for the complete and rapid draining of the entire lubricating system.

2. *Oil Piping*

A. All piping should be of sufficient size to permit smooth flow of oil. Oil piping should have an inside diameter not less than the inside diameter of the engine inlet or outlet. In no case should the suction line from the tank to the pressure pump be less than $\frac{3}{4}$ inch in diameter.
B. Tubing of not less than $\frac{1}{2}$ inch in diameter should be used for the vent line.
C. The drain line should be at least $\frac{1}{2}$ inch in diameter, and larger if possible. The drain pipes should be so located that no discharge will come in contact with any part of the airplane.
D. No oil piping should have splices between connections.
E. The bends in other than flexible tubing should be not less than two diameters in radius.
F. Supports for the oil system are similar to those recommended for the fuel system. In general, the piping should be supported at or near to and on either side of a connection or elbow.

3. *Oil Cooling*

A. Details of oil coolers and regulators are given in the appendix.
B. The lubricating system should be capable of maintaining outlet oil from the engine at a temperature not in excess of 185°F. (85°C.) in an atmosphere of 100°F. (38°C.) when the airplane is climbing at full engine power and full load at sea level.
C. A thermometer is required to measure the temperature of the oil leaving the engine and an additional thermometer to measure the temperature of the oil entering the oil pump.
D. Where an oil cooler is used, means for by-passing the oil cooler, controlled by the pilot or flight mechanic, should be provided.

E. Oil temperature regulators should be installed between the oil tank and the discharge side of the scavenger pump.
F. A pressure gauge for indicating the pressure in the lubricating system should be installed on the discharge side of the pressure pump.

Engine Controls

A. Engine controls consist of all mechanisms exterior to the engine required for controlling spark, throttle and mixture adjustments, as well as cowling shutters control and the like.
B. To open the throttle, move throttle control handle forward.
C. To advance the spark, move spark control handle forward.
D. To lean the mixture, move mixture control handle forward.
E. To open shutters, move shutter control handle forward.
F. To reverse, do the opposite of the operations indicated in B, C, D, and E.
G. Engine controls should be placed at the left of the pilot's seat, except for side-by-side seating of pilots, when they should be located between the pilots' seats.
H. All engine controls should be marked plainly to show their function and method of operation.
I. Throttle control and ignition switches should be easily accessible to the pilot and so arranged as to afford a positive means of controlling all engines separately or simultaneously.
J. A positive means for shutting off all ignition must be readily accessible to the pilot.
K. Throttle controls may be operated by an approved positive-action cable or wire control system.
L. The controls should be positively operated and springs should not be relied upon to actuate the controls in either direction. Push and pull rods should be used wherever possible.
M. Engine controls should be so designed as to avoid the undoing of a large number of bolts or unions to take the rods apart.
N. Ball-bearing supports should be provided wherever possible.
O. Each control system should have one lever of adjustable length.
P. All rod lengths should be adjustable and stiff enough to take 70 pounds on the control handle without failure.
Q. Adjustable stops should be provided.

Electrical Equipment

A. Electrical equipment should be installed in accordance with accepted practice and suitably protected from fuel, oil, water and any other detrimental substance.

B. Adequate clearance should be provided between wiring carrying an appreciable current and fuel or oil tanks, fuel and oil lines, carburetor, exhaust piping and any moving parts.

C. All high tension wiring should be incased in suitable conduits.

D. Ignition circuits should be secured against excessive vibration and adequately protected against short circuits or grounds caused by chafing.
Each engine should be provided with a separate ignition switch. In multi-engined airplanes, the ignition switches should be so placed that all can be readily operated with one hand as well as individually.

E. Fuses should be so located that they can be readily replaced in flight. They must break the current in a generating system at a sufficiently small current flow to protect adequately the lights and other parts of the circuit.

F. Batteries should be easily accessible and adequately isolated from fuel and oil systems.

G. Adjacent parts of the aircraft structure should be protected with a suitable acid-proof paint if the battery contains acid or other corrosive substance.

H. All batteries should be installed so that spilled liquid will be drained suitably or absorbed without coming in contact with the airplane structure.

I. If the battery is completely enclosed, suitable ventilation should be provided.

J. The battery should be mounted so as to be accessible from the outside of the airplane.

K. It should be possible to supply water and test the battery without removing it.

L. Radio units should be accessible and removable from outside of the airplane without entering the cabin.

M. Transmitters, receivers, and dynamators should be easily replaceable.

N. Landing lights should be replaceable at the rear side.

O. Bulbs for all lights should be easily replaceable.

Instruments

The power plant instruments include all those instruments necessary to indicate the behavior of each part. The following are the instruments usually required for each engine.

1. Water thermometer (for water-cooled engines only).
2. Oil thermometer.
3. Oil pressure gauge.
4. Fuel pressure gauge.
5. Tachometer.
6. A manifold-pressure gauge or equivalent for each altitude engine.

Propellers

Some discussion of the determination of the propeller diameter, choice of two-bladed or three-bladed propellers, is contained in the chapter on preliminary performance.

A. Propellers used for commercial aircraft must be approved for the maximum power and speed assigned by the Secretary of Commerce.

B. Propellers may be used on any engine provided that the approved power rating, speed rating and bore of engine (in certain cases maximum engine bore limitations are also assigned to propellers) are not in excess of those assigned to the propeller, and provided that the vibration characteristics of the combination are satisfactory.

C. Fixed or adjustable pitch propellers must be limited to the engine speed at full throttle in level flight at the rated altitude of the engine, not in excess of 105 per cent of the approved rated engine speed. The extra 5 per cent is permitted for the purpose of obtaining additional available power during takeoff and climb.

D. Pending the development of suitable detailed requirements for controllable or automatic-pitch propellers, the controlling mechanism should be so designed that the pilot can prevent the engines from exceeding their rated speed at any time, or their approved output for the flight condition involved.

E. The means provided for the controlling mechanism should be so designed as to minimize the attention required from the pilot in maintaining the proper pitch settings.

F. Propellers should have a minimum ground clearance of 9 inches when the airplane is in a horizontal position with the chassis deflected as it would be under the normal gross weight of the airplane.

G. Propellers on seaplanes should clear the water by at least 18 inches when the seaplane is at rest.

H. A clearance of at least 1 inch should be provided between the tips of the propellers and the fuselage or any part of the structure.

I. Surfaces near the propeller tips should be suitably stiffened against vibration.

J. The pilot and the primary control units, excluding cables and control rods, should not be located in the region between the plane of rotation of any propeller and the surface generated by a line passing through the center of the propeller hub and making an agle of 5 degrees fore or aft of the plane of rotation of the propeller.

K. Likewise, no passenger door should be located in the same region.

Felt

Felt is used quite extensively for a variety of purposes in the airplane such as padding between cowling and attaching structure, shock absorber pads and dust seals for instruments, oil retainers and many other uses. A few of the properties of felt were determined by the author and are given here for possible design use.

1. *Oil Retaining Properties*

For washers or felt pads used to retain oil for lubrication of bearings, shafts, etc., some felts are better than others. A series of oil retaining tests were made by timing the filling of a flask of 60 cc. volumetric capacity with oil (an aviation engine oil of .889 specific gravity was used) which passed through a felt pad placed at the lower end of a 2-inch diameter tube. The average pressure upon the felt was kept constant at 1 lb. per sq. in. and the average time (in seconds) for filling the flask is given in the following table:

TABLE 17—*Oil Retention*

Sample	*Thickness*		
	⅛ in. (*seconds*)	¼ in. (*seconds*)	½ in. (*seconds*)
A	32	50	143
B	39	....	125
C	166	329	463
D	30	78	167
E	108	492	630
F	67	158	310
G	395	....	720

The retention of oil by the felt is dependent upon thickness, density and quality. Sample G (called "Laundry" felt) retained oil best. This is the densest of the felts tested, weighing approximately 4.125 lb. per sq. yd. for a thickness of $\frac{1}{4}$ in. and is useful for lubrication of rocker arms, rotating shafts, etc.

The property of retaining oil is not alone sufficient to determine what felt is best for that particular use.

The oil absorption property is equally important. To determine how fast the felt would absorb oil (of specific gravity .89), specimens of $\frac{1}{4}$ in. x 1 in. x 4 in. were immersed in $\frac{3}{16}$ in. of oil and examined at the end of 20 minutes and at the end of one hour.

TABLE 18—*Oil Absorption*

Sample	*Oil Rise* (20 *min.*) (*inches*)	*Oil Rise* (1 *hour*) (*inches*)	*Weight of Dry Felt* (*grams*)	*Proportion of Its Own Weight of Oil Absorbed*
A	1¼	1⅝	2.95	2.44
B	1¼	1⅝	3.35	1.94
C	1⅛	2	6.80	1.84
D	$1\frac{3}{16}$	$1\frac{5}{16}$	3.20	3.13
E	1	1⅞	7.00	1.83
F	$1\frac{3}{16}$	$1\frac{11}{16}$	5.85	2.91
G	1¼	1¾	4.95	3.19

Another test to determine the oil retaining properties of felt was made by determining the amount of oil seepage past a felt collar around a vertical rotating shaft. A constant weight of oil (25cc.)

was kept above the felt collar so that a constant pressure was maintained. The results are given in Table 19 below:

TABLE 19—*Oil Seepage*

Sample	*Time of Seepage (in seconds)*		
	½ in. Thick	*¼ in. Thick*	*⅛ in. Thick*
A	283	120	67
B	319	140	439
C	808	421	225
D	311	172	127
E	820	634	343
F	406	265	175
G	464	502	608

2. *Wear Tests*

Wherever felt is used to support control cables, wear takes place in the felt. Other cases of wear can be found.

Each specimen, 4 in. square, was loaded at 19.3 grams per sq. in. and then subjected to a linear motion of 180 ft. per min. on a roughened wood surface for various timed intervals. The samples were tested at the same time in order to keep the conditions the same throughout. A coefficient, C, was determined from the following relationship:

$$C_w = W_L/TV$$

where C_w = wear coefficient.

W_L = Loss in weight in grams from original weight of specimen over a certain timed interval.

T = Time in minutes.

V = Velocity of the specimen over the surface in feet per min.

TABLE 20—*Wear Coefficients*

Time (minutes)	*Sample A*	*Sample B*	*Sample G*
10	20.4	23.6	48.4
25	16.7	21.5	47.9
45	12.0	18.4	47.3
60	8.5	16.0	46.7

The wear varied with the quality and the density of the felt.

3. *Resiliency*

The resiliency of felt is most important when felt is used as a chafing strip in supporting gasoline tanks, etc. In such installations it is desirable that the felt deform as little as possible so as to

obviate the necessity of frequent adjustment of supporting straps or replacement due to deformation of the felt under load.

Tests were made on three different grades to determine deformation under load, and permanent deformation. The average results on two samples are given below:

TABLE 21—*Deformation*

Sample	*Original Thickness (inches)*	*Thickness under Load (inches)*	*Deformation (inches)*	*Deformation under Load in % of Original Thickness*	*Permanent Set*	*Permanent Deformation in % of Original Thickness*
E	.500	.275	.225	45	.095	19
D	.490	.140	.350	71.5	.150	30.6
D	.480	.180	.300	62.5	.150	31.3
A	.470	.130	.340	72.3	.170	36.2
A	.475	.125	.350	37.7	.165	34.7

The harder the material, the better the quality of felt, the less the deformation will be.

4. *Coefficient of Friction*

The coefficient of friction is independent of the area and weight, providing the pressure of one surface on the other does not modify the original surface characteristics of the sample tested.

TABLE 22—*Felt Friction Coefficients*

Sample	*Smooth Wood*	*Plate Glass*	*Dry Smooth Metal*	*Oiled Metal Surfaces*			
				I	*II*	*III*	*IV*
A	.352	.220	.198	.176	.178	.259	.259
B	.340	.236	.208	.175	.176	.231	.231
C	.362	.253	.216	.178	.178	.240	.231
F	.382	.272	.222	.178	.178	.240	.222
G	.420	.289	.240	.176	.176	.259	.249

I. "Thin" aviation engine oil applied to smooth surface, felt dry.

II. "Thin" aviation engine oil applied to smooth surface, felt saturated with oil.

III. "Thick" aviation engine oil applied to smooth surface, felt dry.

IV. "Thick" aviation engine oil applied to smooth surface, felt saturated with oil.

The results of these friction tests show only a slight difference in the value of the coefficient of friction between the various grades tested for any particular type of surface. For dry surfaces, the coefficient of friction increases with wool content and density.

For rotating shafts the felt having the lowest coefficient of friction would absorb the least power. This friction coefficient of felt on oiled surfaces is practically independent of the grade of felt used, but dependent upon the viscosity of the oil.

TABLE 23—*Miscellaneous Friction Coefficients*

Surfaces and Materials	*Coefficients*
Leather on Oak	.27 to .38
Leather on Metals, Dry	.56
Leather on Metals, Wet	.36
Leather on Metals, Oily	.15
Woolen Cloth on Woolen Cloth	.435
Hemp on Oak, Wet	.33
Hemp on Oak, Dry	.53

Mark's Handbook shows that the coefficient of friction of felt on wood (dry) is approximately the same as leather on oak (dry); but is less for felt on metal or glass. Leather on oily metal is somewhat lower than for felt on oily metal. Inasmuch as the viscosity of the oil also seems to be a factor, the lower value may be due to the grade of oil used.

TABLE 24—*Identification of Samples*

Sample	*Commercial Name*	*Approximate Lb. Per Sq. Yd. ¼ in. Thick*	*Approximate % Wool Content*
A	Cotton and Wool Pad	2.125	85
B	All Wool Pad	2.125	95-100
C	Wool Back Check	3.750	95-100
D	All Wool Pad	2.125	100
E	All Wool Back Check	4.250	100
F	Soft Laundry	3.000	100
G	Laundry	4.125	100
H	Grey Packing	1.500	35
I	White Athletic Pad	1.250	25
J	Grey Packing	1.750	35

XI. Design of the Wing

In spite of its seeming simplicity, the wing of an airplane requires the most careful study in its design for it is the only part of the airplane which contributes to the lift. A slight change in any one of the factors affecting the wing design is so critical that only one slight difference in two otherwise identical designs would be apparent. These differences might be in performance, such as the top speed or landing speed, or in the rate of climb; again there might be differences in stability—one might be more stable longitudinally, directionally or laterally; or there might be differences in maneuverability.

Pilots who fly externally braced biplanes will attest to the fact that rigging the wings differently by tightening up lift wires or adjusting struts often will change an otherwise beautifully maneuverable airplane to one that is "logy." The change of rigging may have altered the angle of incidence of the wings and therefore the decalage so that entirely different characteristics of the combination are obtained.

From the foregoing statements, it should be apparent that it is not the wing alone that should be considered but its relationship to the rest of the airplane in combination with the fuselage and tail surfaces.

The first three-view, the preliminary weight estimate and the arrangement of the balance diagram are the necessary steps in furnishing the data for the type of wing for the position relative to the fuselage, and for its size. The preliminary three-view has more or less set the shape of the wing and determined whether it is to be a monoplane, a full cantilever or an externally braced wing, or, for example, a biplane with wings of different proportions.

The preliminary weight estimate is instrumental in determining the approximate wing area needed, so that with at least this established, it becomes a comparatively easy matter to select a suitable aspect ratio and thus be able to fix the governing dimensions of span, chord and taper.

The balance diagram is necessary to locate the wing relative to the center of gravity; otherwise difficulty might be experienced later in obtaining suitable static longitudinal stability.

The wing planform may be changed considerably due to certain requirements of landing gear retraction, flap attachments and the like. Suppose it is desired to retract the landing gear straight inboard towards the fuselage without the necessity of swinging it back first and then inboard in order to retract the gear fully into the wing without interfering with the front spar. Such retraction requires that the root portion of the wing be somewhat forward of the leading edge of the mean geometric chord; or, in other words, the wing should have the leading edge swept back so that the root chord will come forward along the fuselage. The reason for this is that there are two original conditions that must be met by the wing and the landing gear. The wing, for example, should be placed so that the 25 per cent point of the mean geometric chord falls directly under the center of gravity. The landing gear on the other hand must be placed at a certain angle ahead of the center of gravity to prevent nosing over. These conditions for the wing and landing gear must be kept, and, unfortunately, these conditions may play havoc with original ideas of wing planform and simple landing gear retraction.

The incorporation of flaps and ailerons often affects wing planform. For simpler operating mechanisms, it may be desirable to have the hinge lines perpendicular to the plane of symmetry of the airplane, or perhaps it may be desirable to have a constant chord flap whose spanwise axis is perpendicular to the plane of symmetry. Both of these more or less arbitrary conditions will affect the ultimate planform of the wing. It is a good plan, therefore, to list at first all the various ideas that the designer wants to incorporate, and then perhaps make preliminary sketches of possible solutions to determine whether the various ideas are compatible.

With this general picture in mind, the embryo designer should now consider the following features of wing design.

Before the design of the wing may be begun it is necessary to study various features which affect its final design. The more important features are considered here although it is impossible to point out all the possible effects of miscellaneous items such as landing lights, engine nacelles, landing gear and fuel tanks.

Airfoil Construction

The ordinates of an airfoil are given in percentages of the chord so that it is an easy matter to determine the depth of a section of a wing when the airfoil and chord length are known.

TABLE 25

Airfoil Section N.A.C.A. 2418 *Chord Length* 10 *inches*

Station	*Station in Per Cent of Chord*	*Station in Inches*	*Upper Ordinate in Per Cent of Chord*	*Upper Ordinate in Inches*	*Lower Ordinate in Per Cent of Chord*	*Lower Ordinate in Inches*
0	0	0.0	0.00	0.00	0.00	0.00
1	1.25	.125	3.28	.328	—2.45	—.245
2	2.50	.250	4.45	.445	—3.44	—.344
3	5.00	.500	6.03	.603	—4.68	—.468
4	7.50	.750	7.17	.717	—5.48	—.548
5	10.	1.00	8.05	.805	—6.03	—.603
6	15.	1.50	9.34	.934	—6.74	—.674
7	20.	2.00	10.15	1.015	—7.09	—.709
8	25.	2.50	10.65	1.065	—7.18	—.718
9	30.	3.00	10.88	1.088	—7.12	—.712
10	40.	4.00	10.71	1.071	—6.71	—.671
11	50.	5.00	9.89	.989	—5.99	—.599
12	60.	6.00	8.65	.865	—5.04	—.504
13	70.	7.00	7.02	.702	—3.97	—.397
14	80.	8.00	5.08	.508	—2.80	—.280
15	90.	9.00	2.81	.281	—1.53	—.153
16	95.	9.50	1.55	.155	— .87	—.087
17	100	10.00	(.19)	.019	(— .19)	—.019
	100	10.00			0.00	—0.00

Table 25 gives the ordinates of the N.A.C.A. 2418 airfoil and the ordinates in inches have been calculated for a 10-inch chord (if the chord were 10 feet, the ordinates would be in feet) and have been plotted as shown in Figure 47. Seventeen stations are usually given.

When the wing employs a different airfoil at the tip than at the root it is possible to determine the ordinates of intermediate wing sections from the known geometric relationships. For example, it is desired to determine the ordinates of a section of a wing, tapered in planform as well as in thickness, a distance x from the root. See Figure 48.

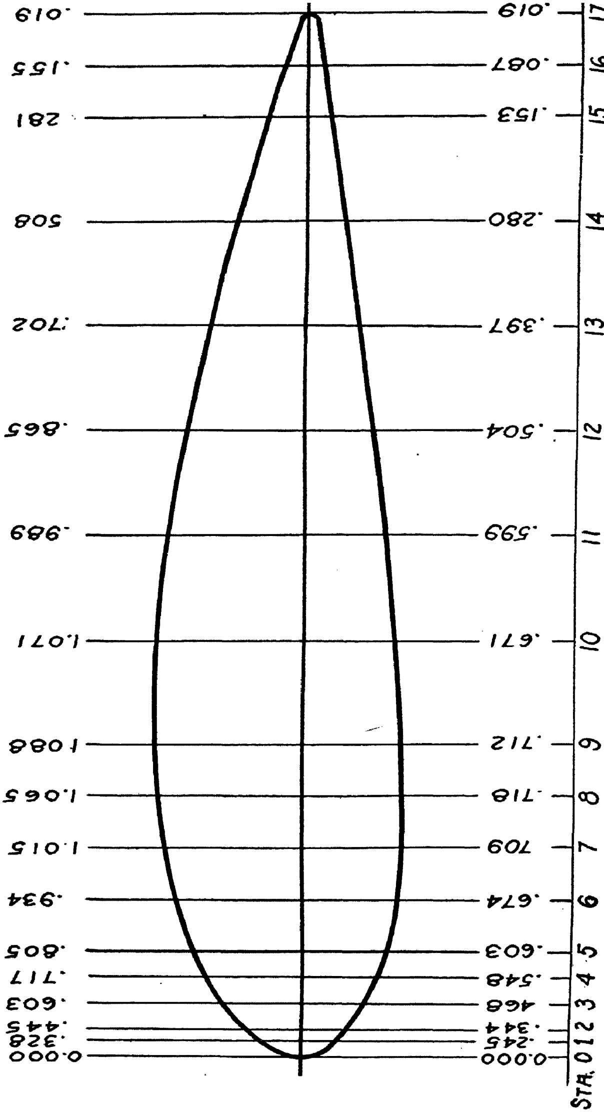


FIGURE 47. ORDINATES OF A 10-INCH AIRFOIL SECTION MEASURED FROM THE CHORD OR DATUM LINE

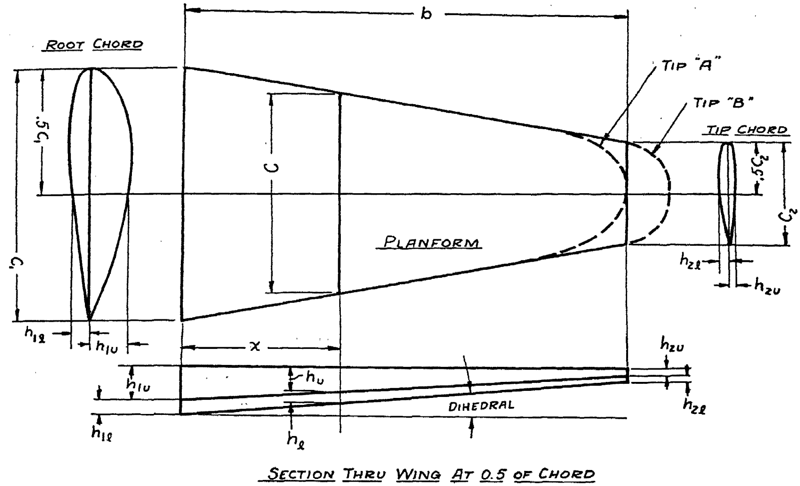


FIGURE 48. A WING WITH DIFFERENT AIRFOILS AT THE ROOT AND AT THE TIP

Corresponding stations are connected by straight lines so that the ordinate for the upper camber at 5/10 of chord for chord

$$\text{C would be } \left[h_{1u} - (h_{1u} - h_{2u}) \quad (x) \right]$$

and the ordinate of the lower camber would be

$$\left[h_{1l} - (h_{1l} - h_{2l}) \quad (x) \right]$$

It is necessary to calculate the ordinate for each station along the chord since the same relationship does not necessarily exist for all the stations except in a few particular cases.

In general it is undesirable to use a radically different airfoil at the tip from that at the root because of the inherent structural difficulties.

Airfoil Selection

In selecting a suitable airfoil for a given design it is necessary to consider the structural and aerodynamic characteristics required for the airplane. A racing airplane, for example, will have an entirely different wing from a transport airplane.

A full cantilever wing, with an aspect ratio from 6 to 8, requires a thickness ratio of at least 18 per cent for the root section. The larger the aspect ratio, the greater the thickness ratio of the root section should be. It is not desirable to use the same thickness ratio from root to tip, but to decrease it linearly to not less than 9 per cent at the tip.

A semi-cantilever wing, with an aspect ratio from 6 to 8, requires a thickness ratio of at least 12 per cent for the root section. Again, the larger the aspect ratio for the wing, the greater the thickness ratio of the root section must be. The wing should be decreased linearly to about 6 per cent thickness ratio for the tip chord.

A biplane would probably use an airfoil of thickness ratio from 6 to 9 per cent and use the same airfoil from root to tip.

When the thickness ratio for the airfoil has been tentatively established, it is then desirable to compare a group of airfoils to determine which has the best all-around characteristics. Table 26 below shows the comparison of representative airfoils that have been selected.

The characteristics investigated throw light on the eventual performance of the airplane using the airfoil chosen. The maximum lift coefficient, $C_{L_{max}}$ should be as high as possible since, for the same landing speed, a smaller wing area will be required. The lift over drag ratio, $\frac{L}{D} = \frac{C_L}{C_D}$ (which is the ratio of the lift coefficient to the drag coefficient at the same angle of attack) gives an indication of the cruising radius and speed. The higher this ratio, the more efficient the airplane will be at cruising.

The ratio of $\frac{C_L^{\frac{3}{2}}}{C_D}$ (again C_L and C_D are values for same angle of attack) is known as the power coefficient and should also be calculated. The best rate of climb is usually obtained when this ratio has its minimum value, and the lower the value of this ratio, the greater is the rate of climb. Other ratios are sometimes determined as the speed range ratio $\frac{C_{D_{min}}}{C_{L_{max}}}$ for example, where $C_{L_{max}}$ is indication of the minimum speed and $C_{D_{min}}$ an indication of the maximum speed.

In addition to the aerodynamic characteristics just enumerated, the depths of the airfoil at certain percentage distances along the

TABLE 26—*Airfoil Characteristics*

Section	Test by Lab.	Aspect Ratio R	Reynold's Number	Slope of Lift Curve $\frac{dC_L}{d\alpha}$	α (Deg) for $C_L = 0$	C_M at $C_L = 0$	CP Most Forward	$C_{L_{max}}$	α (Deg) for $C_{L_{max}}$	$C_{D_{min}}$	α (Deg) for $C_{D_{min}}$	C_L/C_D	Depth at 15%	Depth at 20%	Depth at 60%	Depth at 70%
NACA 0006	LMAL	6	3.21×10^6	0.102	—0.1	—0.002	0.25	0.88	13	0.0065	0	23.8	5.35	5.74	4.56	3.66
" 0009	"	6	3.21×10^6	0.101	0.00	—0.003	0.25	1.27	14	0.0074	—1	22.9	8.02	8.60	6.84	5.49
" 0012	"	6	3.23×10^6	0.101	0.00	—0.002	0.25	1.53	17	0.0083	—1	22.2	10.69	11.47	9.12	7.32
" 0015	"	6	3.2×10^6	0.100	0.00	—0.000	0.25	1.53	17	0.0093	0	21.2	13.36	14.34	11.40	9.16
" 0018	"	6	3.15×10^6	0.098	0.00	—0.002	0.24	1.49	17	0.0108	0	19.8	16.04	17.20	13.69	10.99
" 2509	"	6	3.06×10^6	0.102	—2.0	—0.052	0.29	1.38	13	0.0083	—1	22.9	8.04	8.62	6.85	5.50
" 2512	"	6	3.08×10^6	0.102	—2.1	—0.054	0.28	1.62	17	0.0091	—1	22.3	10.70	11.46	9.12	7.32
" 2515	"	6	3.08×10^6	0.099	—2.0	—0.049	0.28	1.53	16	0.0104	—1	20.4	13.38	14.35	11.40	9.16
" 2518	"	6	3.09×10^6	0.096	—2.0	—0.047	0.28	1.48	16	0.0113	—1	19.1	16.07	17.25	13.69	10.99
" 2521	"	6	3.13×10^6	0.095	—1.8	—0.043	0.28	1.38	16	0.0126	—1	17.7	18.72	20.09	15.97	12.84
" 21012	"	6	8.37×10^6	...	—0.6	—0.001	0.25	1.67	15	0.0070	0	22.0	10.69	11.47	9.13	7.33
" 22012	"	6	8.32×10^6	...	—0.9	—0.005	0.25	1.66	16	0.0073	0	22.8	10.69	11.49	9.12	7.33
" 23012	"	6	8.48×10^6	...	—1.2	—0.005	0.26	1.67	16	0.0070	0	23.0	10.69	11.47	9.14	7.36
" 24012	"	6	8.26×10^6	...	—1.5	—0.013	0.26	1.65	16	0.0073	—1	22.3	10.71	11.48	9.14	7.32
" 25012	"	6	8.24×10^6	...	—1.6	—0.019	0.27	1.61	15	0.0077	—1	21.9	10.72	11.51	9.14	7.34
" 22112	"	6	8.55×10^6	...	—0.8	—0.001	0.25	1.58	15	0.0073	0	22.1	10.69	11.48	9.13	7.34
" 23112	"	6	8.21×10^6	...	—0.8	0.002	0.25	1.67	16	0.0075	—1	22.3	10.69	11.48	9.15	7.31
" 24112	"	6	8.00×10^6	...	—0.9	0.00	0.25	1.61	16	0.0075	—1	22.3	10.71	11.47	9.13	7.32
" 25112	"	6	8.24×10^6	...	—1.2	0.002	0.27	1.56	15	0.0075	0	22.3	10.73	11.48	9.13	7.33
Clark Y	"	6	3.6×10^6	0.071	—5.1	—0.081	0.294	1.370	16.0	0.0106	—5.5	21.0	10.51	11.31	...	7.34
G 387	"	6	3.6×10^6	0.072	—6.8	—0.096	0.301	1.329	14.6	0.0126	—6.0	18.6	13.83	14.71	...	9.23
M 6	"	6	3.6×10^6	0.070	—0.2	—0.010	...	1.222	18.6	0.0080	—0.0	21.9	10.29	11.17	...	8.06
M 12	"	6	3.6×10^6	0.071	—1.2	—0.005	0.246	1.293	18.6	0.0089	—1.5	20.6	10.21	11.10	...	7.98
RAF 15	"	6	3.6×10^6	0.072	—2.2	—0.052	0.289	1.211	15.7	0.0083	—1.5	24.3	6.38	6.27	5.24	4.79
USA 27	"	6	3.6×10^6	0.070	—4.6	—0.090	0.293	1.386	16.7	0.0116	—4.0	20.6	10.60	11.73	9.82	8.14
USA 35A	"	6	3.6×10^6	0.071	—7.9	—0.119	0.342	1.208	13.5	0.0142	—6.7	18.4	16.60	17.65	13.17	10.51
USA 35B	"	6	3.6×10^6	0.072	—5.1	—0.072	0.302	1.377	16.0	0.0093	—4.8	20.5	10.56	11.23	8.36	6.66

chord are important to know for structural reasons. A too-shallow depth for a cantilever spar for instance will give too heavy a structure for the strength required.

A useful table of comparison follows:

Airfoil Section	
Tested at	
Aspect Ratio	
Reynold's Number	
Slope of Lift Curve	
Angle of Attack for $C_L = O$	
Angle of Attack for $C_{L_{max}}$	Determines angle of airplane, for three-point landing
Maximum C_L	Indicative of Wing Area Required
Minimum C_D	Indicative of Top Speed
Angle of Attack for Minimum C_D	Indicative of Wing Incidence
Maximum $\frac{L}{D}$	Indicative of Cruising Speed
Angle of Attack at Maximum $\frac{L}{D}$	Indicative of Cruising Angle
$\frac{L}{D}$ at $\frac{2}{3}$ Maximum C_L	For Good Climb and Cruising Characteristics
$\frac{L}{D}$ at $\frac{1}{2}$ Maximum C_L	
$\frac{L}{D}$ at $\frac{1}{4}$ Maximum C_L	For Transport Work
$\frac{L}{D}$ at $\frac{1}{6}$ Maximum C_L	For Moderately High Speed Airplanes

$\frac{L}{D}$ at $\frac{1}{8}$ Maximum C_L	For High Speed Airplanes
Minimum Value of $C_L^{3/2}/C_D$	For Best Climb
$C_{L_{max}}/C_{D_{min}}$	Speed Range
Spar Depth at 15, 20, 60 and 70%	The Deeper the Spar the Lighter the Structure
Most Forward Position of Center of Pressure at Max. C_L	Smallest movement gives most efficient structure.
Most Rearward Position of Center of Pressure at Min. C_D	

Aspect Ratio Corrections

With few exceptions, wind tunnel tests of airfoils are made for a standard aspect ratio of 6, but since the wing used for the design very seldom has just an aspect ratio of 6, it is necessary to correct the airfoil characteristics for the proper aspect ratio.

The aspect ratio of a wing is determined from the ratio

$$\frac{(\text{Span})^2}{\text{Area}} = \frac{b^2}{S}$$

The span is measured from tip to tip. The area of the wing includes that area covered by the fuselage and shown as the shaded portion in Figure 50.

It should be noted here that the area covered by the fuselage is used only for aspect ratio calculations. For all subsequent calculations, such as effective area for calculation of landing speed or in performance calculations, it is *not* included unless it is unobstructed both above and below as for a parasol monoplane.

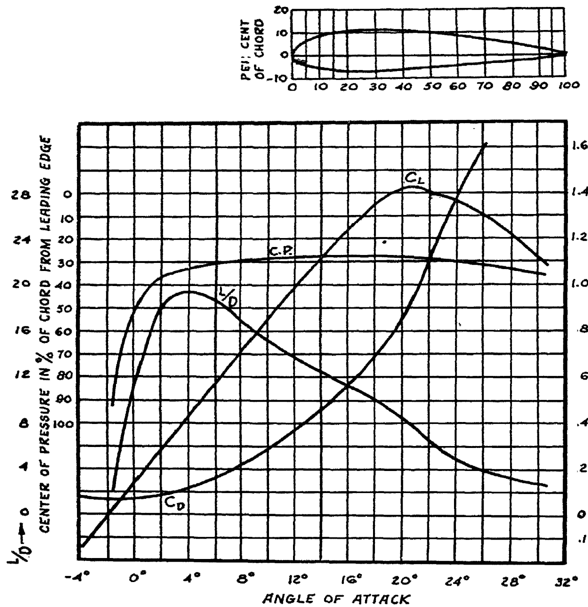
The drag coefficient may then be corrected for the new aspect ratio by means of the formula

$$C_D = C_{D_o} + C_{D_i} = C_{D_o} + \frac{C_L^2}{\pi R}$$

where C_D is the total drag coefficient for the airfoil used.

C_{D_o} is the profile drag and is independent of the aspect ratio, and constant for the airfoil

$C_{D_i} = \frac{C_L^2}{\pi R}$ is the induced drag.



%	UP'R	L'W'R
0	0.00	0.00
1.25	3.28	-2.45
2.50	4.45	-3.44
5.00	6.03	-4.68
7.50	7.17	-5.48
10	8.05	-6.03
15	9.34	-6.74
20	10.15	-7.09
25	10.65	-7.18
30	10.88	-7.12
40	10.71	-6.71
50	9.89	-5.99
60	8.65	-5.04
70	7.02	-3.97
80	5.08	-2.80
90	2.81	-1.53
95	1.55	-0.87
100	(.19)	(-.19)
100	—	0.00

L.E. RAD: 3.549
SLOPE OF RADIUS THROUGH END OF CHORD: 2/20

NAME OF SECTION: N.A.C.A. 2418
SIZE OF MODEL: 5" × 30"
PRESSURE IN STANDARD ATMOSPHERES: 20.8
WIND VELOCITY: 68.6 FT/SEC.
RESULTS CORRECTED TO ASPECT RATIO 6 IN FREE AIR

REYNOLDS NO: 3,060,000
TEST: V.D.T. 669
DATE: SEPT. 11, 1931

FIGURE 49. TYPICAL REPRESENTATION OF AERODYNAMIC CHARACTERISTICS OF AN AIRFOIL

To calculate the drag coefficient, for example, for aspect ratio 8 when the characteristics for aspect ratio 6 are known, the following formula may be derived

$$C_{D_R} = C_{D_6} + \frac{C_L^2}{\pi}\left[\frac{1}{R} - \frac{1}{6}\right] \quad \text{where } R = 8$$

Corrections for low values of C_L may be ignored. The angle of attack must likewise be corrected for aspect ratio

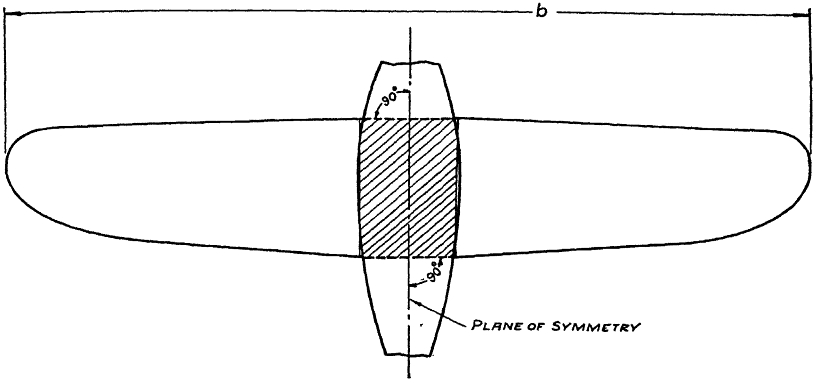


FIGURE 50. FOR ASPECT RATIO CALCULATIONS, THE SHADED PORTION, REPRESENTING THE WING AREA COVERED BY THE FUSELAGE, IS INCLUDED IN THE TOTAL AREA

$$\alpha = \alpha_o + \frac{C_L}{\pi R}$$

$$\text{or } \alpha_R = \alpha_6 + 57.3\ C_L \left[\frac{1}{R} - \frac{1}{6}\right]$$

when the known characteristics are for aspect ratio 6. The lift coefficient, the center of pressure and the corrected drag coefficient correspond to the corrected angle of attack. Table 27 has been set up to expedite the aspect ratio calculations. The various terms are self-explanatory.

When an airfoil section is used for the tip of a wing different from that used at the root, it is necessary to make wind tunnel tests on a model of the actual wing. Reasonably close approximation for preliminary calculations may be obtained by averaging the characteristics of the root and tip airfoils.

Mean Geometric Chord

The center of gravity of the complete airplane is placed, usually, at the maximum forward position of the center of pressure on the mean aerodynamic chord in order to get the desired stability.

TABLE 27—*Computation of Airfoil Characteristics*

1. C_L	-1.0	-.8	-.6	-.4	-.2 0	.2	.4	.6	.8	1.0	1.2	1.4	1.6	1.8	2.0	2.2
2. α_6																
3. $\Delta \alpha = 18.24\ K\ C_L$																
4. $\alpha = (2) + (3)$																
5. C_{D_6}																
6. $\Delta C_{Di} = .318\ K\ C_L^2$																
7. $C_D = (5) + (6)$																
8. COS α = COS (4)																
9. SIN α = SIN (4)																
10. C_L COS α = (1) x (8)																
11. C_D SIN α = (7) x (9)																
12. $C_N = (10) + (11)$																
13. C.P. = C.P.$_6$																
14. $C_{M_c/4} = (.25 - (13)) \times (12)$																
15. $C_{M_a} = (14) + (a - .25) \times (12)$																
16. $C_{D_i} = (6)/KR$																
17. $C_{D_o} = (7) - (16)$																
18. ΔC_{L_u} (BIPLANE)																
19. ΔC_L (BIPLANE)																
20. $C_{L_u} = (1) + (18)$																
21. $C_{L_l} = (1) + (19)$																

R = $K = \frac{1}{R} - \frac{1}{6}$

$\frac{57.3}{\quad} = 18.24$

The mean aerodynamic chord is difficult to determine unless the pressure distribution for the wing being designed is definitely known. Moreover, the pressure distribution varies with the angle of attack. It is therefore customary to use the mean geometric chord of the wing instead.

The mean aerodynamic chord or the mean geometric chord is determined for only one half of the wing either up to the side of the fuselage for a wing whose center portion is blanketed by the fuselage, or up to the plane of symmetry as in the case of a parasol monoplane.

The mean geometric chord of a rectangular wing is just half of the semi-span out from the fuselage as shown in Figure 51.

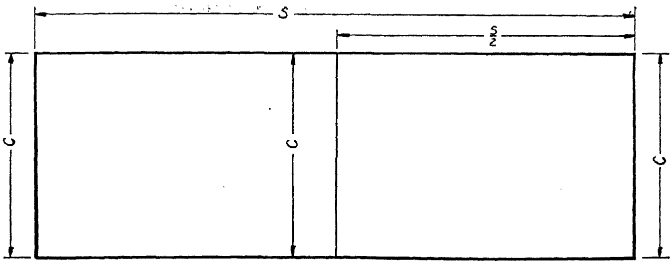


FIGURE 51

If the tip of the wing is tapered or rounded off, an equivalent semi-span can be determined such that the area included in the equivalent semi-span is equal to the area excluded. This is shown in Figure 52 A.

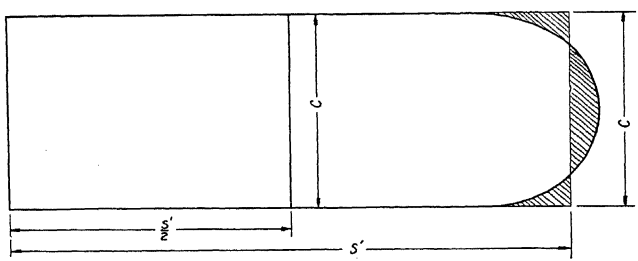


FIGURE 52 A

for a rectangular wing and in Figure 52 B for a tapered wing.

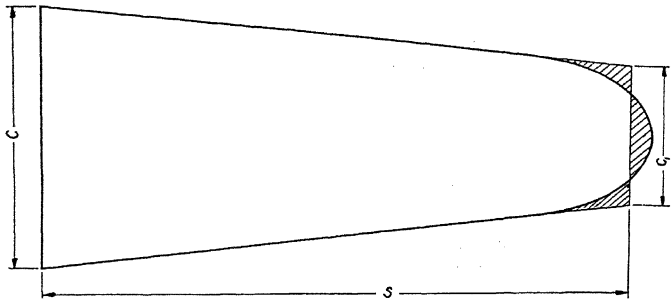


FIGURE 52 B

A geometrical method for finding the mean geometric chord is shown in Figure 53.

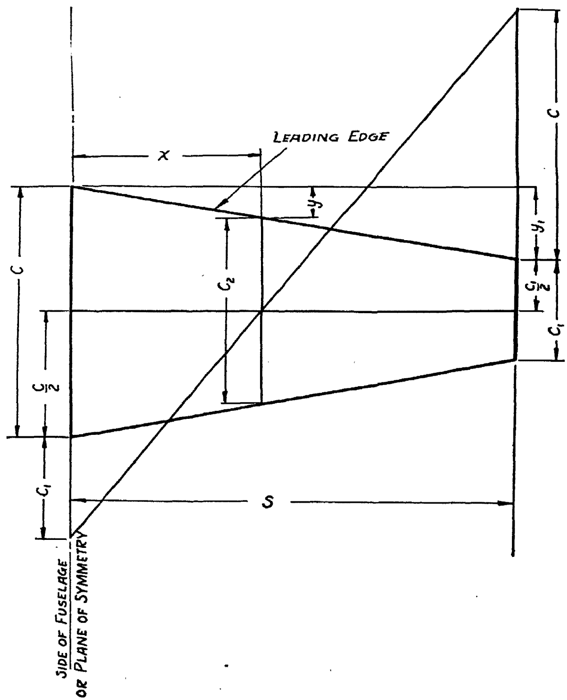


FIGURE 53

The length of the mean geometric chord may also be calculated by means of the formula

$$C_2 = \frac{2}{3}\left[C + C_1 - \frac{C\,C_1}{C + C_1}\right]$$

where the various terms are as indicated in Figure 53.

The distance out from the side of the fuselage or plane of symmetry may be calculated from

$$X = \frac{S\ (C + 2\ C_1)}{3\ (C + C_1)}$$

and the distance in rear of the leading edge of the root chord.

$$y = \frac{y_1\ (C + 2\ C_1)}{3\ (C + C_1)}$$

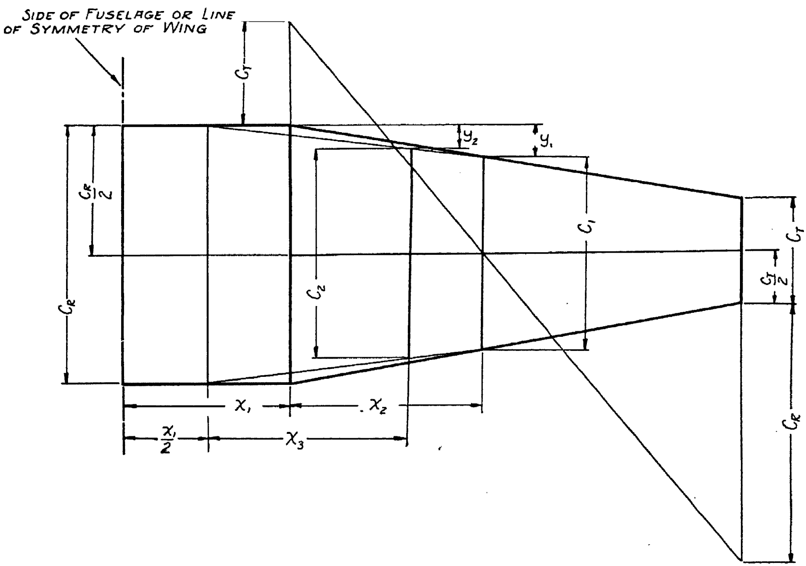


FIGURE 54

When the wing is made up of a rectangular and a tapered portion, the procedure, referring to Figure 54, is to

A. Determine the mean geometric chord of the trapezoidol and the rectangular section separately.

B. If the area of the rectangle is A, and the trapezoid A_2 then the mean geometric chord of the combination is

$$C_2 = \frac{C_R\ A_1 + C_1\ A_2}{A_1 + A_2}$$

Its distance from the mean geometric chord of the rectangular section is

$$X_3 = A_2\ \left(X_2 + \frac{X_1}{2}\right) \div (A_1 + A_2).$$

If the root section is not rectangular, the method is similar.

The mean aerodynamic chord may be determined also by the elemental strip method as shown in Figure 55. The location of the chord out from the side of the fuselage or plane of symmetry may be expressed mathematically:

$$\overline{X} = \frac{\int C_L x ds}{\int C_L dS} = \frac{\int C_L x C dx}{\int C_L C dx} = \frac{\Sigma [C_L x C]}{\Sigma [C_L C]}$$

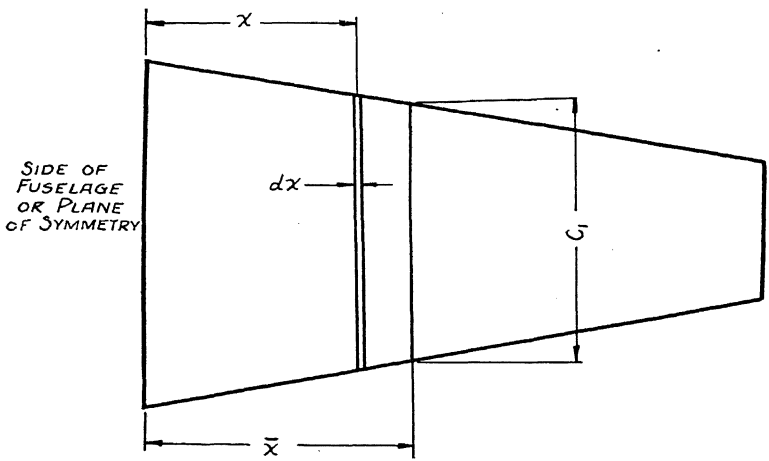


FIGURE 55

The integration is best done mathematically by choosing an elemental strip of width dx = unity. If the airfoil used is the same from root to tip, and the angle of incidence of each strip is the same, C_L will cancel in the numerator and the denominator. C is the chord at the distance x from the reference plane.

The mean geometric chord of a biplane combination is slightly more involved. Referring to Figure 56

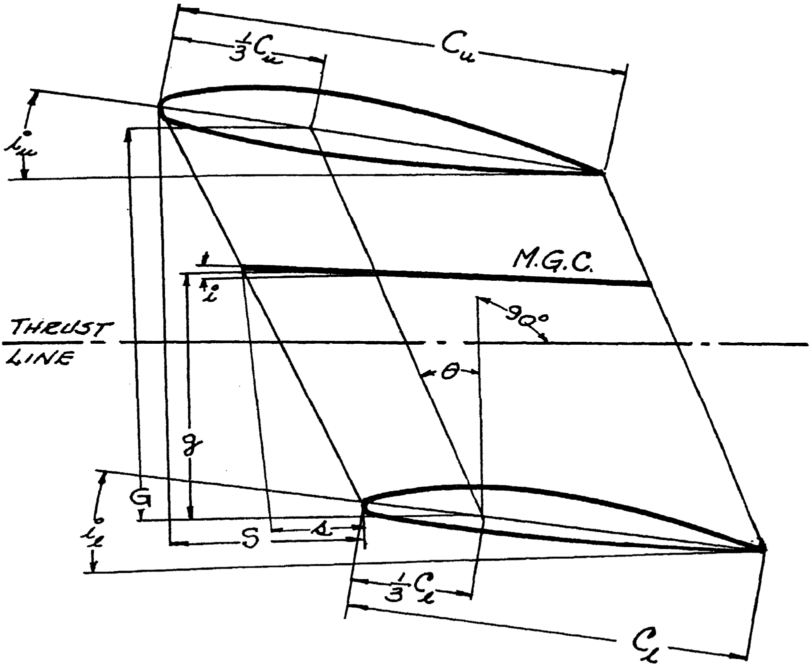


FIGURE 56

C_u = Mean geometric chord of upper wing in feet.
A_u = area in square feet of upper wing.
i_u = angle of incidence of upper wing.
C_l = Mean geometric chord of lower wing.
A_l = area in square feet of lower wing.
i_l = angle of incidence of lower wing.
e = relative efficiency of upper to lower wing.
G = gap in feet.
O = angle of stagger.

The relative efficiency, e, of the upper to the lower wing, depends upon the airfoil or airfoils used, angles of decalage, dihedral, and stagger, as well as the gap-mean chord ratio, but sufficiently close approximation may be made by determining e from Figure 57.

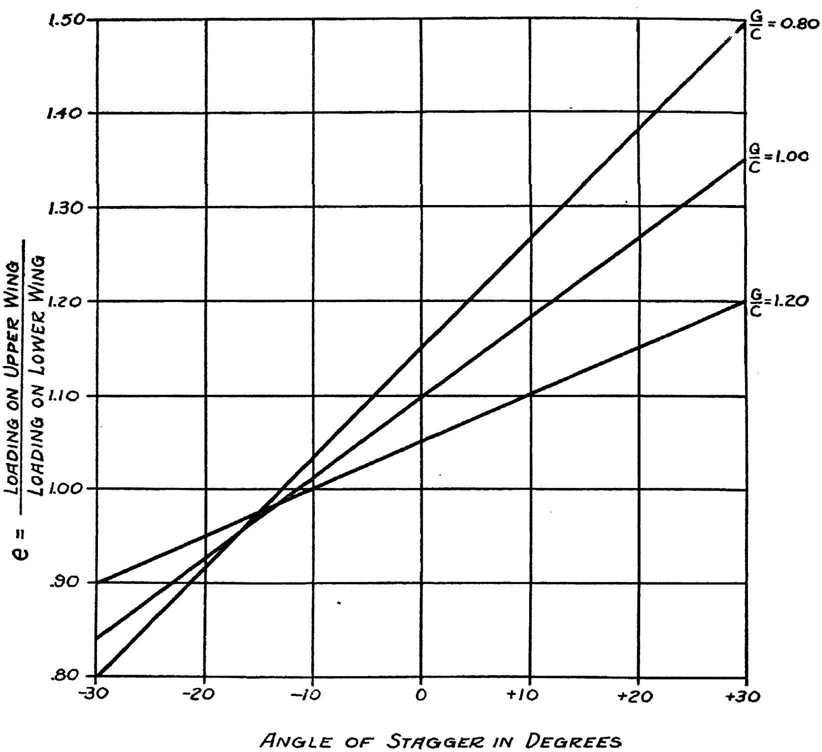


FIGURE 57. CORRECTIONS FOR DETERMINING THE MEAN AERODYNAMIC CHORD OF A BIPLANE COMBINATION

The mean geometric chord for the biplane is then

$$C = \frac{e\ C_u A_u + C_l\ A_l}{e\ A_u + A_l}$$

$$g = \frac{Ge\ A_u}{eA_u + A_l} = \frac{G\ (C - C_l)}{C_u - C_l}$$

$$s = \frac{Sg}{G}$$

Wing Planform

Wing planforms vary according to the whim of the designer. However, a few standard designs are more or less used. Cantilever wings of high aspect ratio—for example, use a taper ratio of 2 to 1

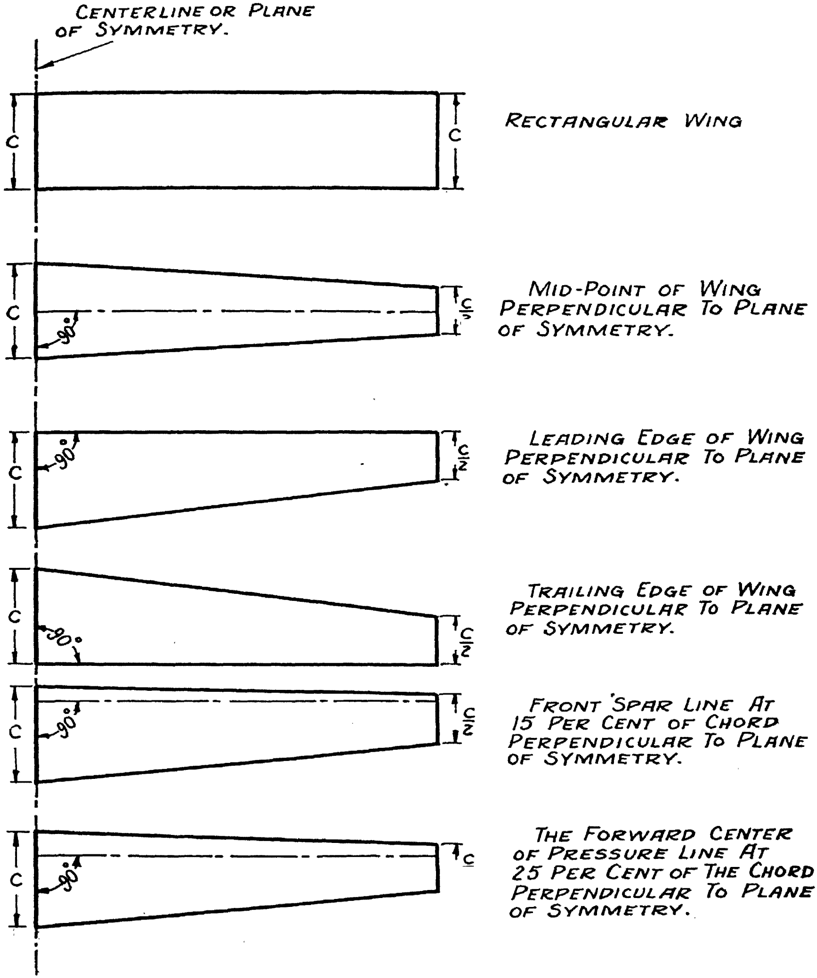


FIGURE 58. VARIATIONS IN WING PLANFORMS

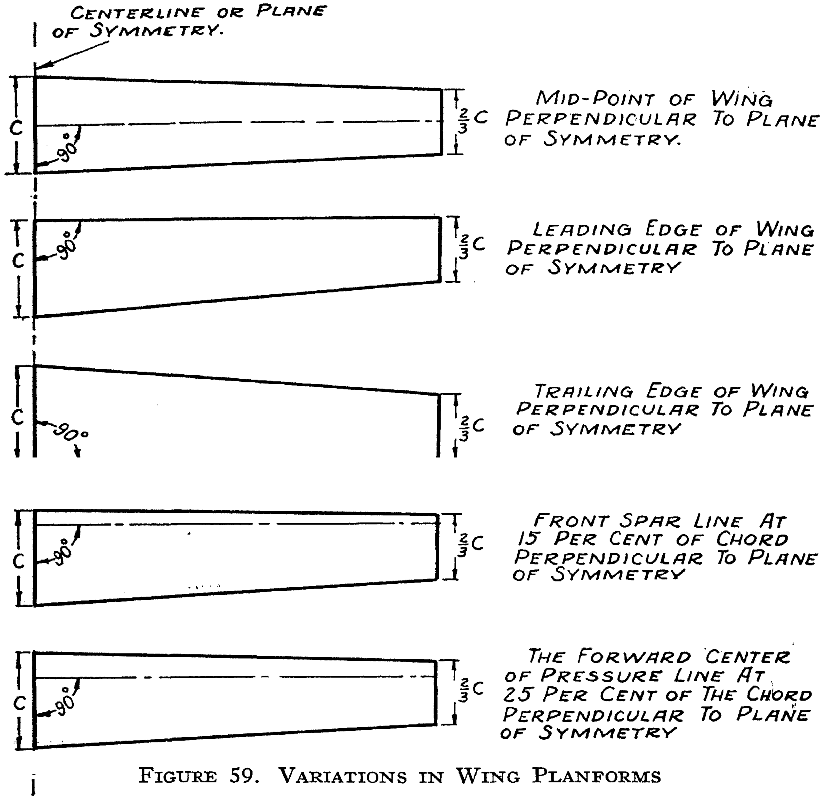


FIGURE 59. VARIATIONS IN WING PLANFORMS

(to as high as 4 to 1)—while wings of not so high aspect ratios, relatively, use a taper ratio of 3 to 2 of the root to the tip chord.

In addition to the taper ratio, additional variation is obtained by sweeping the leading edge rearward depending upon the effect desired. Figures 58 and 59 show some of these variations. It is not always desirable to fix upon the planform at once since the location of the mean geometric chord and the spar locations are often the determining factors.

Figure 60 shows a straight taper while Figure 61 includes a rectangular section alongside of the fuselage. The latter is often desirable for fuel tank placement and attachment of landing gear, particularly of the retracting type.

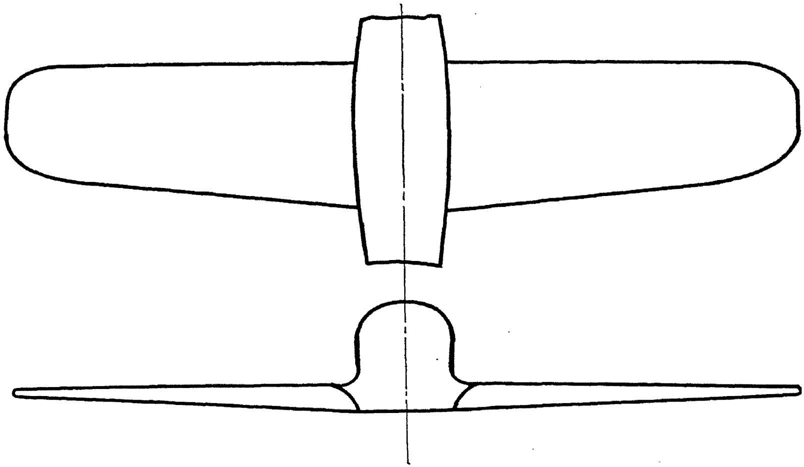


FIGURE 60. A WING TAPERING IN PLANFORM SHOWN IN CONJUNCTION WITH THE FUSELAGE

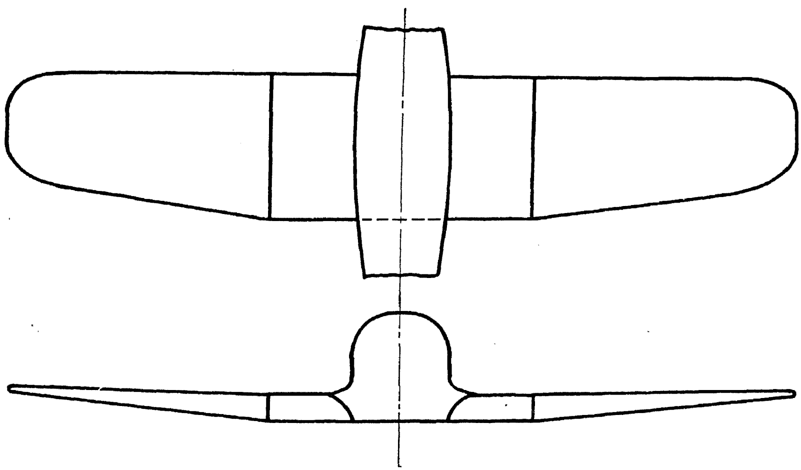


FIGURE 61. A RECTANGULAR SECTION AT THE ROOT AND A TAPERED SECTION OUTBOARD GIVES THIS EFFECT TO THE PLANFORM OF A WING

The wing tip shape is determined more by the designers than by aerodynamic considerations. A few likely wing tip forms are shown in Figure 62.

Further discussion of wing tip forms is given under aileron design.

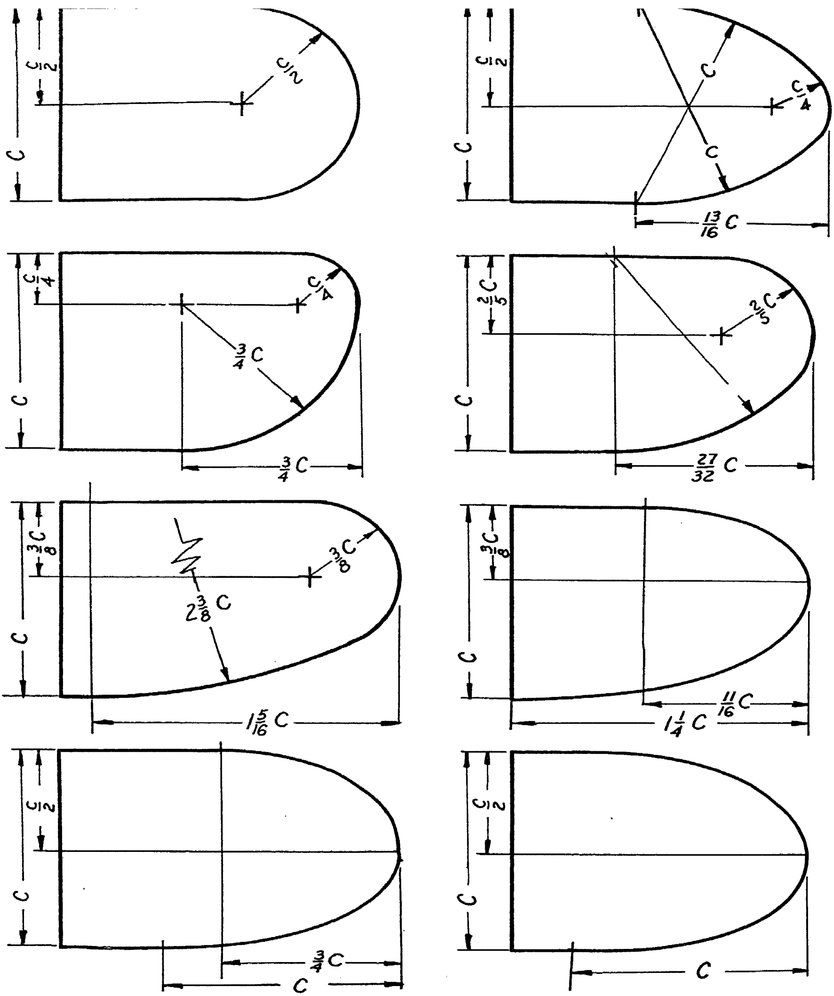


FIGURE 62. SUGGESTED TIP PLANFORMS FOR WINGS

Dihedral

The amount of dihedral that must be given to a wing must ultimately be determined by wind tunnel tests.

A wing equipped with flaps will probably require more dihedral than one without since the span is relatively shorter for the former than for the latter. However, it is not a question of lateral stability alone but of the relation of lateral to directional stability as well.

It is customary to give at least 3 degrees up to as much as 6 degrees dihedral to the wing. This may be given to the entire wing from the fuselage outward or from the stub wing outward. Again in the latter case more dihedral would be required than for the former case.

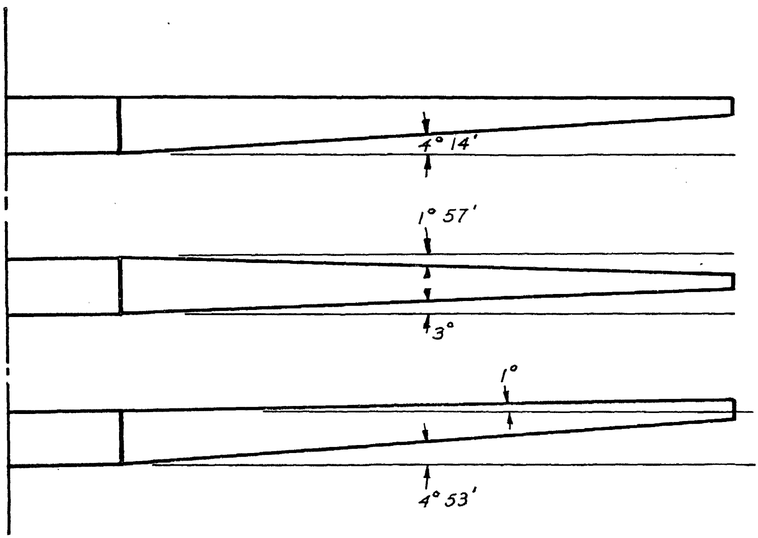


FIGURE 63. THE EFFECT OF DIHEDRAL ON THE FRONT VIEW OF A WING

Figure 63 shows the effect of keeping the top surface of the wing horizontal; or of allowing only 3 degrees dihedral; or allowing 1 degree for the top surface. The first and third case are more desirable from the point of appearance.

It is difficult to determine what the effective dihedral is for a wing tapered in planform and thickness. It is easiest, as well as custom-

ary, to measure the dihedral angle to the bottom surface. It is sometimes measured to the plane made by the chord lines of the individual airfoil sections.

Fixed Angle of Wing Setting

Theoretically, an airplane designed for high speed should have its wing set at such an angle to the fuselage that the combination will give the least possible drag. This setting is difficult to determine without wind tunnel tests due to unknown interference effects.

A good compromise is to set the wing at an angle to the longitudinal axis of the fuselage corresponding to the angle at which minimum drag occurs.

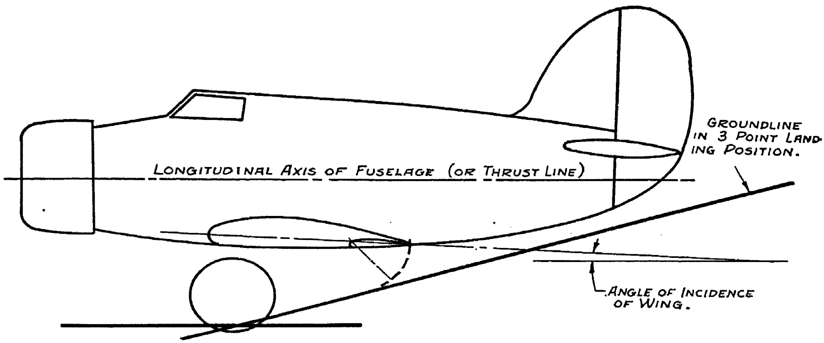


FIGURE 64. WING SETTING AND FLAP CLEARANCES

Other considerations may be the deciding factors such as flap clearance in the deflected position, or the inconvenient ground angle for the fuselage when in the three-point landing position. The former case usually requires a smaller fixed angle of wing setting while the latter requires a larger angle of incidence.

Ailerons

Figure 65 shows a few possible planforms for ailerons, labeled as to whether a particular planform is desirable or not. Figure A of this group shows an aileron which diminishes in effectiveness at high angles of attack, although very effective at low angles. Figure B shows the aileron slightly too far inboard which is particularly undesirable when the maximum span for a flap as a lift increase device is required.

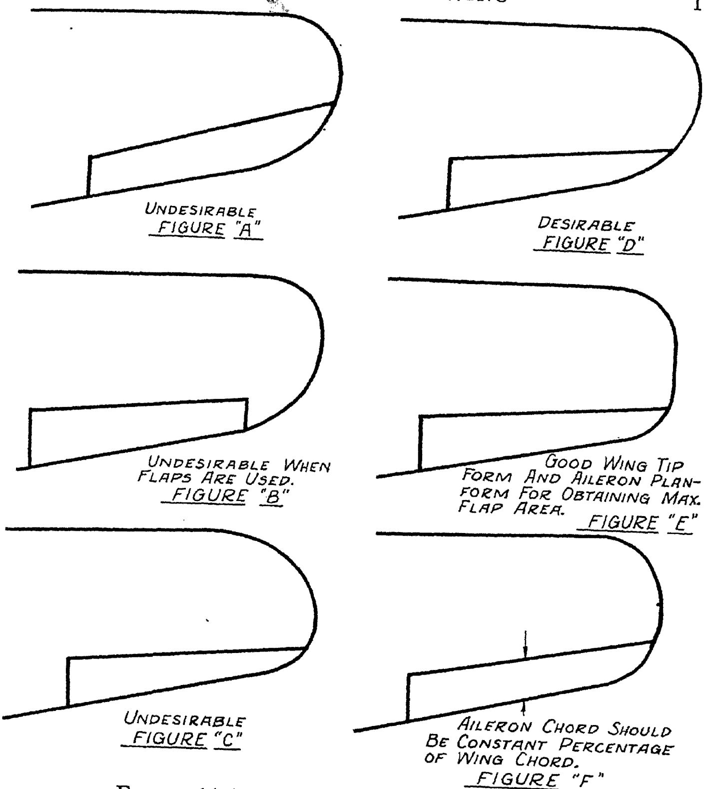


FIGURE 65 A POSSIBLE AILERON PLANFORMS

Figure C is a bad combination of a relatively good aileron and a bad wing tip in that the same effect is obtained as for Figure "A."

Figures D, E and F show generally desirable planforms. An aileron preferably should be not more than 25 per cent of the chord although 30 per cent is common when flaps are used for increased lift.

The next question that arises is whether the aileron should be aerodynamically balanced. For relatively slow speed airplanes and possibly for high speed airplanes utilizing auxiliary mechanical, electrical or hydraulic aid, unbalanced ailerons may be used.

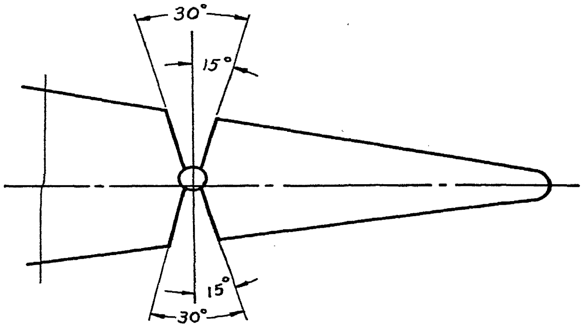


FIGURE 65 B. A SIMPLE UNBALANCED AILERON

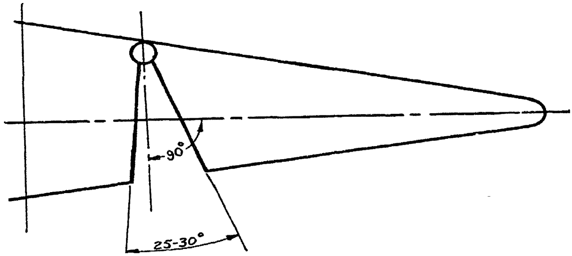


FIGURE 65 C. ANOTHER FORM OF AN UNBALANCED AILERON

Figures 65 A and B show two standard forms of these unbalanced ailerons.

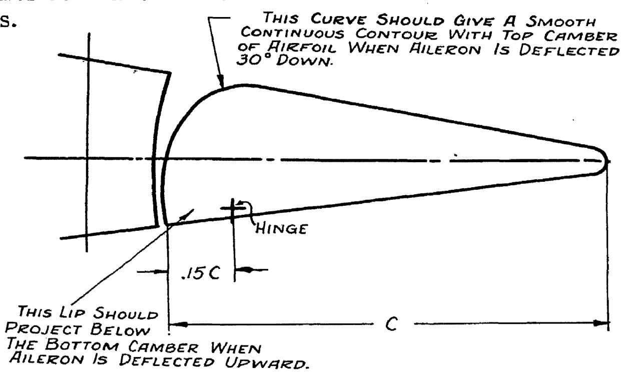


FIGURE 66 A. A BALANCED AILERON WHICH IS GOOD AERODYNAMICALLY, BUT MAY COLLECT ICE UNDER ICING CONDITIONS

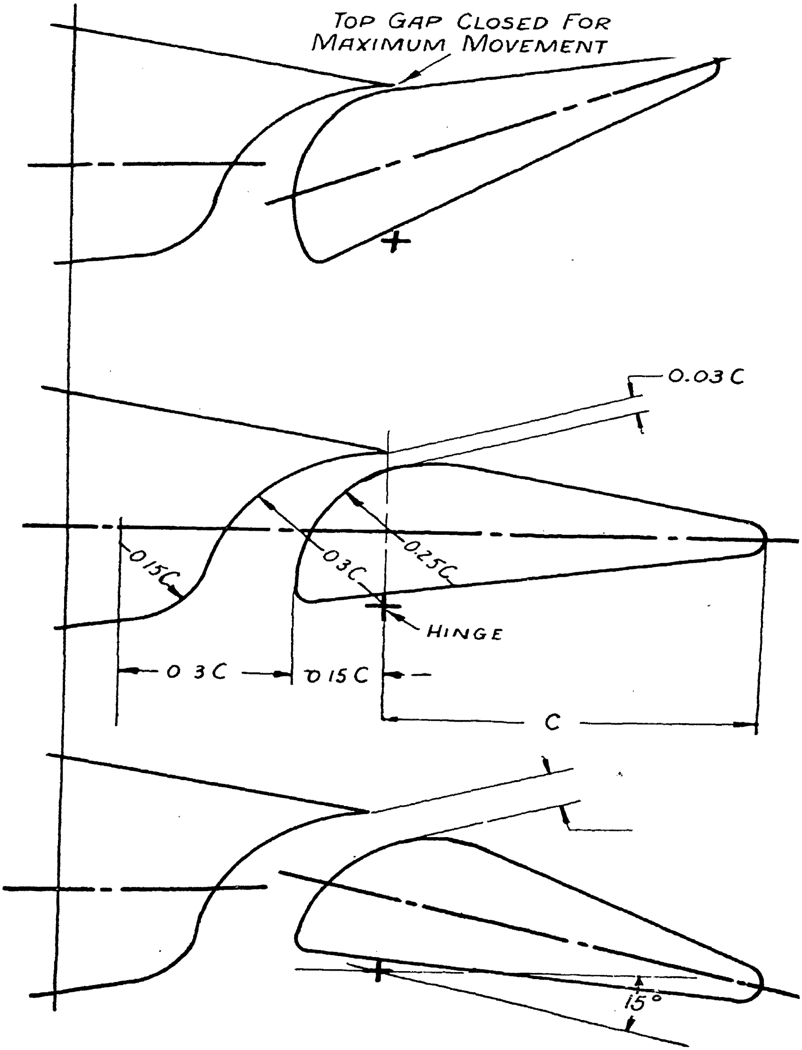


FIGURE 66 B. ANOTHER BALANCED AILERON WHICH HAS BEEN FOUND EFFECTIVE EVEN UNDER ICING CONDITIONS

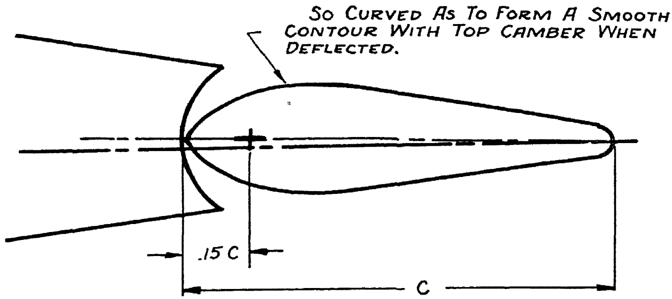


FIGURE 66 C. THIS BALANCED AILERON HAS NO SHARP PROJECTIONS IN ITS DEFLECTED POSITIONS

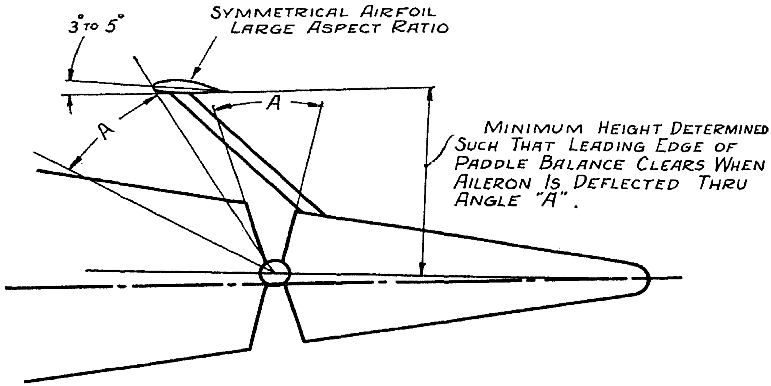


FIGURE 66 D. A BALANCED AILERON SUITABLE FOR MODERATELY HIGH SPEED AIRPLANES

For high speed and large airplanes, aerodynamically balanced ailerons are shown in Figures 66 A, B, C, and D.

In order to reduce the manual effort to operate ailerons, a type of servo-control has been devised whereby a small surface is deflected in order to move a larger surface. Another feature affecting aileron design is possible icing effects. The leading edge preferably should not project beyond the normal contour of the wing when deflected, since ice may form on the leading edge of the control surface.

For control purposes, these trailing edge flaps or tabs should have as high an aspect ratio as possible and about 10 to 12 per cent of the total movable area. A smaller percentage is sufficient when the tab is intended for trim purposes. Reference to the chapter on tail surfaces should be made for further explanations.

Aileron areas vary from 8 to 12 per cent of the total wing area (including the aileron area which is considered as the wing area).

Lift Increase Devices

To attain a high top speed it is desirable to have a high wing loading, but unless some lift increase device is used, the landing speed may be well beyond the desirable limit.

Various lift increase devices have been invented, and a few likely ones are shown in Figure 67 A.

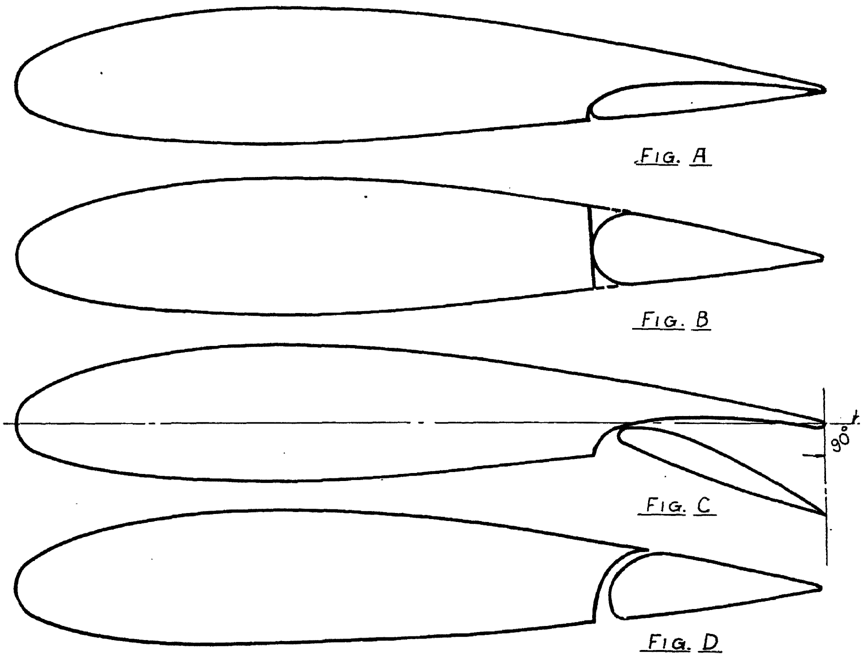


FIGURE 67 A. TYPICAL LIFT INCREASE DEVICES
The flaps shown in Figures A, B and D move downward angularly. The flap in Figure C moves rearward also with the locus of its trailing edge as shown.

Flaps of the type shown in Figure 67 A will give approximately 70 per cent increase in the maximum lift coefficient for the wing area affected. Figure 67 B shows a Fowler type of flap which is moved

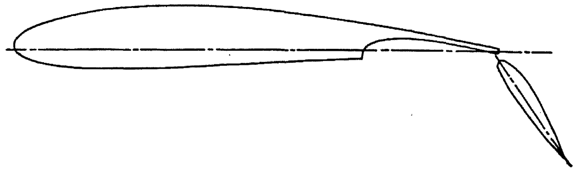


FIGURE 67 B. THE FOWLER TYPE FLAP WHICH MOVES REARWARD AND DOWNWARD

not only angularly but rearwardly also. This type of flap will give a 100 per cent increase in the lift over the basic wing area. For all types of flaps it usually is customary to deflect the flap not more than 45 degrees, and to make the flap chord about 30 per cent of the wing chord so that attachments may be made to the rear spar in a two spar system.

For an airplane equipped with conventional size ailerons, the maximum lift coefficient for the entire wing area is often arbitrarily assumed to have been increased 40 per cent.

A more analytical method is to calculate the "effective" maximum lift coefficient. Referring to Figure 68 a suggested method would be

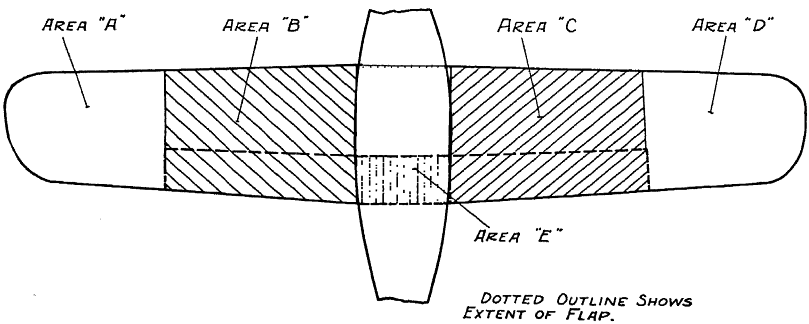


FIGURE 68

$$\left[(C_{L_{max}}) \text{(Areas A + D)} + (1.7\ C_{L_{max}}) \text{(Areas B + C)} + (0.1\ C_{L_{max}}) \text{(Area E)} \right] \div \left[\text{Areas (A + B + C + D)} \right] = \text{The Effective } C_{L_{max}}$$

The area covered by the fuselage is normally not included in the wing area, but if a flap is made continuous across the bottom of the fuselage, additional lift results depending upon the type of fuselage. This addition has been estimated at 1/10 of the maximum lift coefficient in the above empirical formula.

Structure

The structure of the wing is dependent upon its final arrangement on the location of the aileron, the flap, the fuel tanks, the retractible landing gear and any other items peculiar to the particular design.

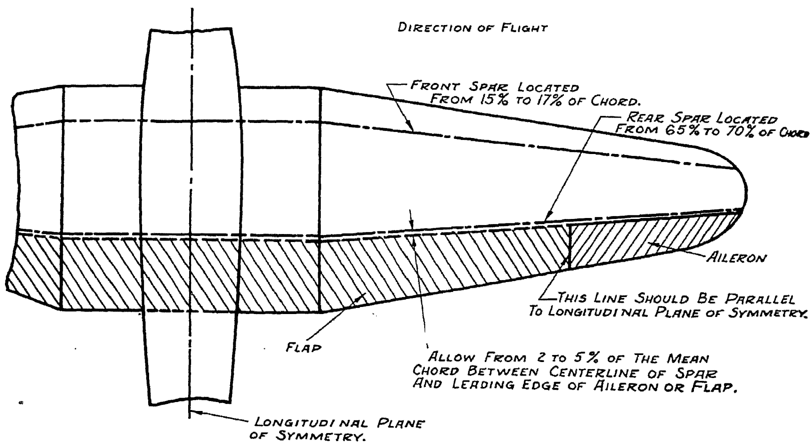


FIGURE 69. SUGGESTIONS FOR LOCATING SPARS, AILERONS AND FLAPS ON A WING

Figure 69 shows the allocation of area of the aileron and flap, and the suggested location of the front and rear spar.

Spar construction varies. A few typical metal spars are shown in

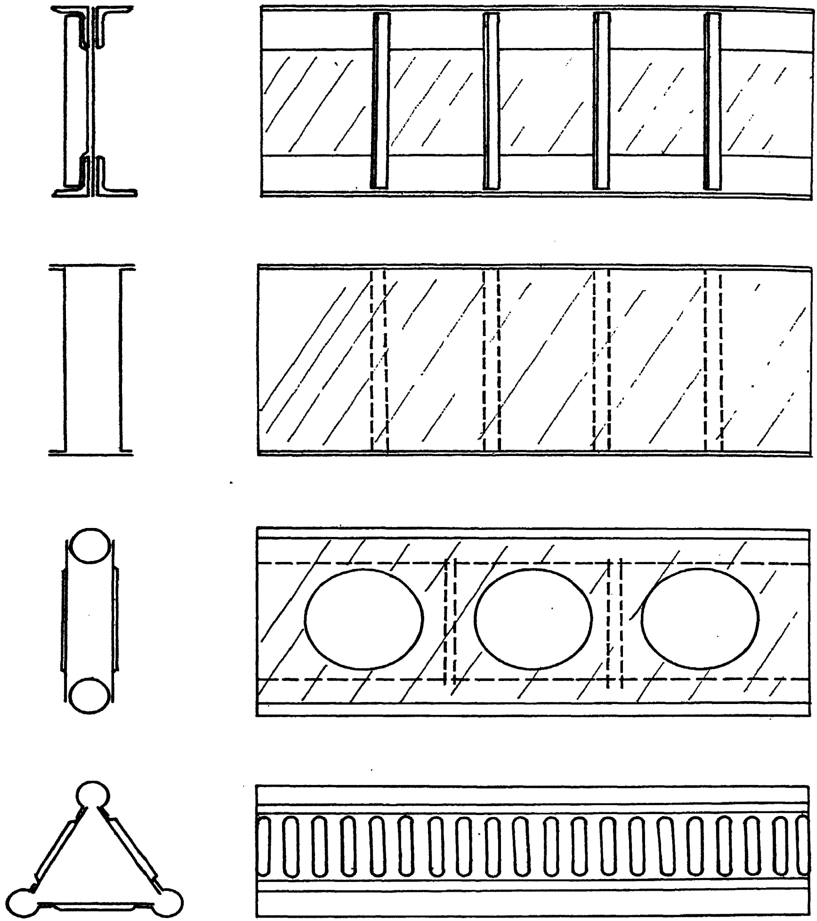


FIGURE 70 A. SPARS OF METAL CONSTRUCTION
The top spar is an example of the tension field type spar.

Figures 70 A and B. There are any number of combinations of these types possible and probably all have been used at some time or another.

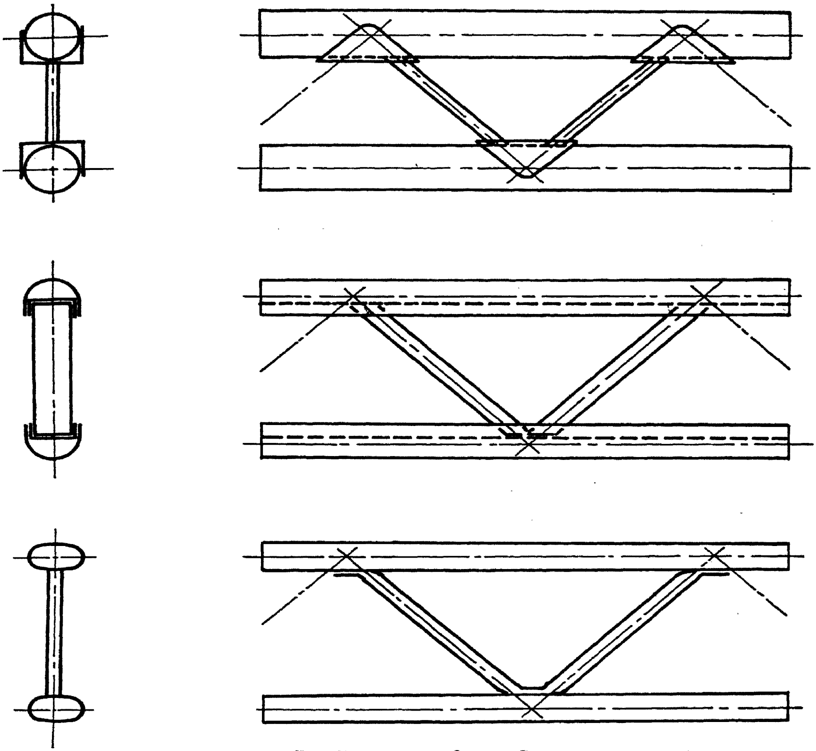


FIGURE 70 B. POSSIBLE SPAR STRUCTURES

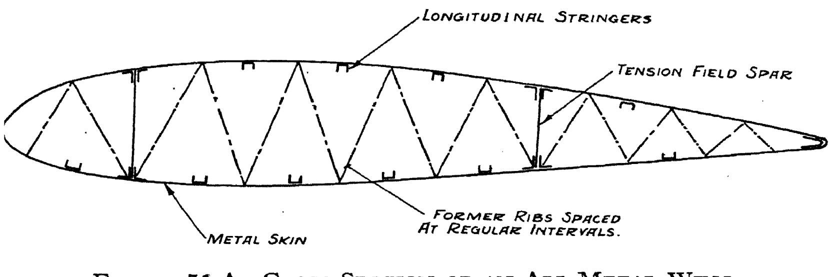


FIGURE 71 A. CROSS SECTION OF AN ALL-METAL WING WITH TENSION FIELD SPARS

Sections of typical wing construction are shown in Figures 71 A and B. It is desirable to study illustrations in leading aeronautical magazines to get a conception of possible aircraft structures.

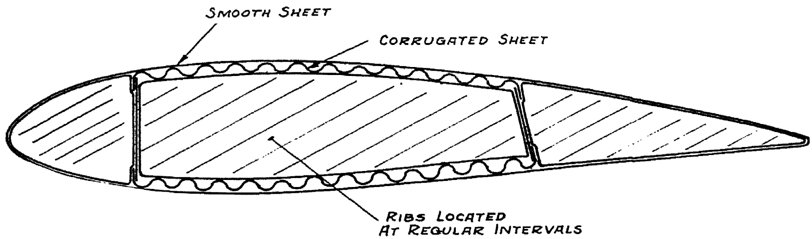


FIGURE 71 B. CROSS SECTION OF AN ALL-METAL WING

Flutter Prevention

Precautions must be taken in the design of the wing to prevent flutter. The Department of Commerce recommends:

(*a*) structural stiffness.
(*b*) elimination of all play in hinges and control system joints.
(*c*) rigid interconnection between ailerons.
(*d*) a relatively high degree of weight balance of control surfaces.
(*e*) a relatively low amount of aerodynamic balance.
(*f*) high frictional damping.
(*g*) adequate fillets.
(*h*) proper fairing.

Features tending to create aerodynamic disturbance, such as sharp leading edges on movable surfaces, should be avoided; these are also prone to collect ice.

When ailerons are attached to internally braced wings, they should be statically balanced about their hinge lines. If a balancing weight is used, it is usually more effective when located near the outboard end of the aileron.

Partial static balance may be satisfactory when an irreversible and exceptionally rigid control system is used.

Additional information on the prevention of flutter may be found in the chapter on tail surfaces.

PROBLEMS

1. Lay out the planform of the semi-span of a wing which has a total area of 450 square feet for the complete span. The planform of the semi-span is trapezoidal; the ratio of the root chord to the tip chord is 3 to 2; and the aspect ratio of the wing is to be 6.5.

2. What are the ordinates of the airfoil section at the mid-section of an outer wing panel of length 20 feet; root chord 12 feet; tip chord 8 feet; if the root chord employs the N.A.C.A. 2418 airfoil and the tip chord employs the N.A.C.A. 2409 airfoil?

3. Determine the aspect ratio of a wing with the following dimensions: root chord, 8 ft.; tip chord, 6 ft.; width of fuselage, 4 ft. Total span 60 feet.

XII. The Landing Gear

The landing gear consists of the wheels, tires, brakes, shock-absorbers, struts, cowlings and, if retractable, the retracting mechanism. With only minor exceptions, most of the items making up the landing gear are not designed by the airplane designer but by the accessory manufacturer. However, their selection and their relationship are determined by the airplane designer so that it is important for him to know the various conditions which the landing gear must meet and the purpose for which each part has been designed.

The landing gear must, of course, take the shocks when landing or when going over an obstruction, and so it incorporates two means of absorbing shock loads—the tire which absorbs minor shocks, and the shock absorber which absorbs hard and sudden shocks. But not only must the landing gear be able to take these shocks, it must also be so placed that upon landing the airplane will be prevented from nosing over. In order to accomplish this, the front wheels of the conventional landing gear are placed somewhat ahead of the center of gravity of the airplane.

While it may be desirable to have the landing gear reasonably far enough ahead of the center of gravity of the airplane, yet if placed too far forward there would be difficulty in taking off. In taking off, the tail of the airplane must be raised until the longitudinal axis of the airplane is practically horizontal. In this position, the airplane accelerates quickly until it reaches a climbing speed and is ready to take off. But in order to reach this horizontal attitude, there must be a lift on the horizontal tail surfaces produced by the relative wind on these tail surfaces caused by the propeller slipstream and the forward acceleration of the craft. This lift multiplied by the distance from the center of pressure on the horizontal tail surfaces to the point of contact of the wheel with the ground is the moment which must be just equal to the moment produced by the weight of the airplane times the distance from the center of gravity of the airplane to the point of wheel and ground contact. When these two moments are equal, the airplane will be in a horizontal attitude. As soon as the airplane starts to accelerate, the elevators, which have been depressed up to this time, are gradually neutralized; otherwise too much lift would be created and the airplane would nose over.

If the front wheels were quite far ahead of the center of gravity of the airplane, a greater moment would have to be produced by the horizontal tail surfaces. Since the lift on the tail surfaces is proportional to the square of the speed, it would be necessary to increase the speed to obtain the necessary lift. However, it takes time to start at zero speed and accelerate up to a particular speed, and the longer it takes to accelerate, the longer will be the take-off run. The recommendations given in this chapter are the result of many years of experience of aircraft designers and present the best all-around compromise of all factors involved.

Brakes are used to reduce the landing run. If the brakes were used immediately upon level landing, the inertia of the airplane might be sufficient to nose it over. It is therefore necessary to put the wheels farther forward for a landing gear employing brakes than one without brakes.

Landing introduces another problem in the disposition of the wheels. If the tail wheel is too close to the front wheels, or the front wheels are too close together in relation to the span of the wings, the airplane may ground-loop—a phenomenon in which the airplane may pivot on one wheel, meanwhile dragging a wing tip along the ground.

The present interest is in the so-called "tri-cycle" landing gear. This reverses the location of the single wheel which in the conventional landing gear is the tail wheel and now becomes the nose wheel for the new type.

Whereas the center of gravity was slightly *behind* the two wheels, it is now slightly *ahead* of the two wheels, and the nose wheel is placed as far forward as the particular airplane design will permit.

Various tests have been made by the Army, by the National Advisory Committee for Aeronautics, and by a few commercial companies. The results of these tests and investigations may be summarized as follows:

1. There should be greater passenger comfort since the passengers are sitting in a cabin which is level not only when flying but also on the take-off, and in landing. With conventional gear of today, the three-point attitude of the airplane on landing and at the start of the take-off may be decidedly uncomfortable.
2. There is better vision both for the pilot and for the passengers not only in landing but also on the take-off.
3. The tri-cycle landing gear gives greater ground stability since the three wheels are likely to be more evenly loaded at all times.

4. There should be no tendency to nose over since the nose wheel, being ahead of the center of gravity of the airplane, would resist any nosing over. The nose wheel is therefore a definite nosing-over preventative.
5. Since the airplane cannot nose over, there is the possibility of landing at almost any angle of attack. For the private flyer who may be a "dub" pilot, this is a very good feature since the landing technique need not be well-nigh letter perfect. For transport airplanes, the possibility of landing at almost any angle is advantageous in blind flying when the ground is not visible.
6. Also because nosing over is unlikely, it is possible to have a shorter landing run since the brakes can be applied as soon as contact with the ground is made. Moreover, since the lift on the wings is less at the moment of landing due to the smaller angle of attack, the load on the wheels will be greater, and with brakes on all three wheels, the braking will become more effective.
7. The smaller angle of incidence of the airplane with the ground will permit the airplane to accelerate faster up to take-off speed and practically "automatic" take-offs are possible and in much shorter time than for the conventional landing gear.
8. The quicker take-offs and shorter landing runs permit a shorter block-to-block speed which is an important factor in economical commercial air transportation.
9. The airplane rests on wheels with the wings at a smaller angle of attack than for the conventional landing gear. The lift coefficient of the wings at small angles of attack is small so that even at relatively large wind speeds, the lift on the wings is not likely to be great enough to blow the airplane over.

Against these very favorable advantages of the tri-cycle landing gear, there should be balanced these possible disadvantages which may be entirely or at least partially overcome by proper design:

1. If the two main wheels in rear of the center of gravity are too far back, the load on the nose wheel will be increased. This will necessitate a heavier nose wheel.
2. If the two main wheels are too near the center of gravity, the nose wheel will not have enough load on it and will therefore tend to bounce more easily when taxying.
3. If the front wheel is located too close to the two main wheels. one of the following may occur:
   a. A sudden swerve of the airplane may be followed by turning over about a line connecting the nose wheel with one of the two main wheels.
   b. The front or nose wheel may "shimmy" unless there is friction damping.

   c. The airplane may have a tendency to "buck."
4. Unless the airplane can assume a greater angle of attack at take-off, the take-off run on muddy ground may be long. The propeller thrust and the higher ground drag due to the mud seem to cause the nose wheel to dig in at low angles of attack. By raising the nose of the airplane the load on the nose wheel is relieved.
5. With increase in propeller thrust, the load on the nose wheel increases since the propeller thrust line is above the nose wheel. Unless the nose wheel is equipped with a larger "oleo" travel and a stronger structure, severe shock loads are likely to be transmitted to the airplane structure, for the "oleo" or shock absorber travel may be compressed long before any serious loads are imposed.
6. Difficulty may be encountered with the nose wheel in riding over obstacles. The tail wheel seems to behave better under such circumstances.
7. The tail wheel has the advantage of protecting the tail surfaces. Unless a skid or special "crash pad" is provided for the rear portion of the fuselage when the nose wheel type of landing gear is used, the rear portion of the fuselage may be damaged in case of an unusual "tail low" landing.
8. The nose wheel is harder to retract because of its location in the forward portion of the fuselage and because of its longer shock absorber travel.

Wheel and Tire Size

The size of the wheels and tires in the conventional landing gear are determined by the static weight equal to half the gross weight of the airplane per wheel. It is not necessary to determine the load factors and loads imposed by various landing conditions since the wheels and tires are originally designed with ample margins of safety.

The size of the wheels for a tri-cycle landing gear depends upon their position relative to the center of gravity. The two rear wheels may have from 85 to almost 100 per cent of the load while the front wheel may have from 10 to 25 per cent of the gross weight of the airplane as the static load.

Size of Tail Wheel

The weight of the tail wheel for preliminary weight estimate and balance determination can be estimated by assuming a static load of about 1/10 to 1/12 the gross weight of the airplane and then choosing the required tail wheel.

After the center of gravity has been found, the weight and size

of the tail wheel may be corrected by finding the correct static weight on the tail wheel as follows:

$$\text{Static Weight } R_1 = \left[\frac{a}{a+b}\right]\left[\text{Gross Weight}\right]$$

where *a* and *b* are dimensions as indicated in Figure 73.

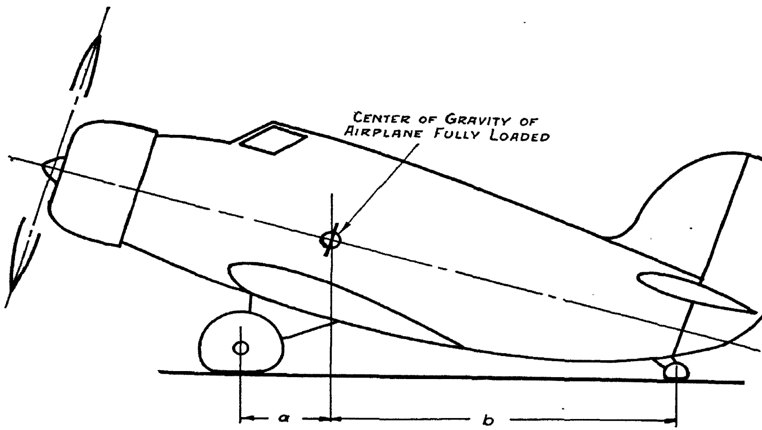


FIGURE 73

Tread

The wheel tread is a function of (a) the height of the center of gravity above the ground, (b) the wing span and (c) the distance between the front wheels and the tail wheel. As these dimensions increase in size, the larger the wheel tread should be for good ground behavior.

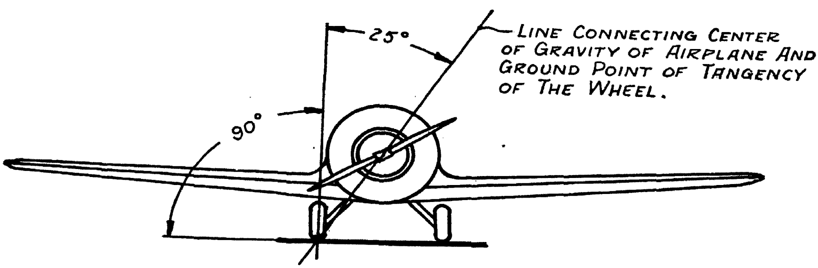


FIGURE 74 A. WHEEL TREAD FOR A SMALL AIRPLANE

Figure 74 A shows a rule to follow for small airplanes carrying not more than 3 or 4 passengers.

Figure 74 B shows a rule to follow for large airplanes of the present day.

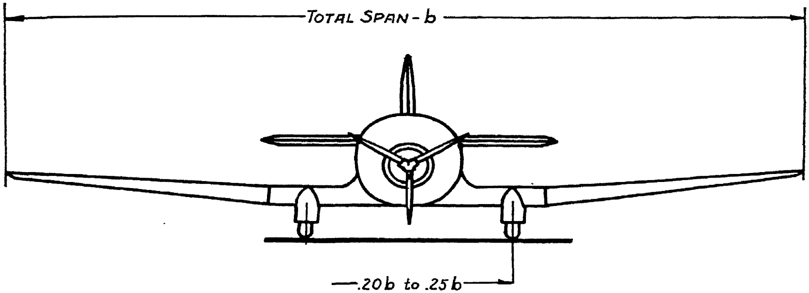


FIGURE 74 B. RECOMMENDED WHEEL TREAD FOR A LARGE AIRPLANE

Wheel Position

Examination of the landing gears reveals that the wheel, without any load on it, as incorporated in the normal landing gear, may "toe-in" or the vertical centerline when viewed from the front is at an angle of several degrees from the vertical—commonly known as "camber." These two features are of course accentuated when the landing gear is in the fully extended position due to the configuration of the particular members of a landing gear. The condition for which to design is the static loaded condition either in the level-landing position or the three-point landing position.

The wheels are given no "toe-in" for the normal condition unless the configuration of the landing gear should be such as to cause an appreciable "toeing" out in the fully contracted position.

The camber given the wheels may be 1 or 2 degrees outward unless, again, the configuration of the landing gear is such as to give an undesirable camber when in the fully contracted position. Too much leeway either way may cause the tire to "roll" off when landing.

The position of the wheels with relation to the center of gravity is shown for the conventional landing gear and the tri-cycle landing gear in Figures 75 A and B.

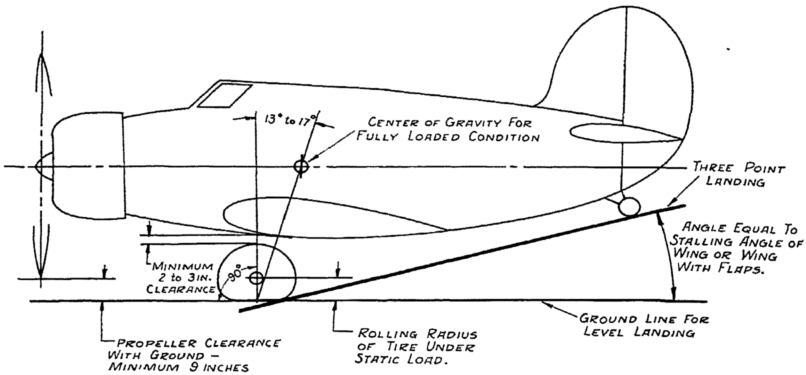


FIGURE 75 A

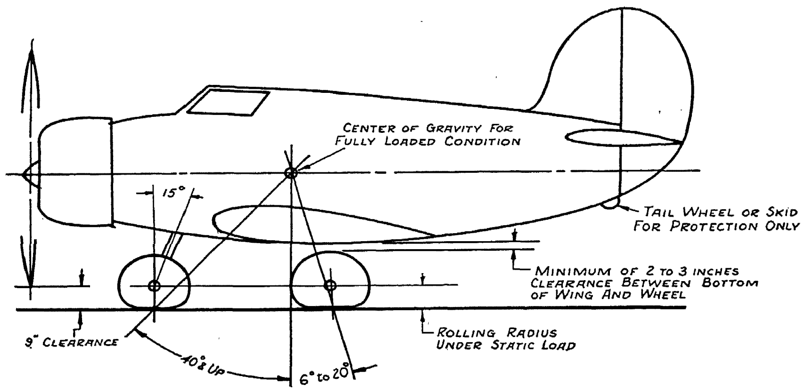


FIGURE 75 B

Shock Absorbers

All landing gears must be equipped with some shock-absorbing device in order to avoid sudden shocks to the passengers, and to avoid the enormously high stresses that would be caused even in a normal landing if no shock absorbers were used.

There are various types of shock absorbers. At the present time, the popular type of shock absorber is one that utilizes an oil which is made to pass from one chamber to another through one or more orifices.

Shock absorbers usually have to be designed for a particular airplane although it is possible to obtain some stock sizes in the lower ranges.

These shock absorbers are designed to absorb the energy created by a free drop, in inches, of 0.36 times the calculated stalling speed in miles per hour (but not greater than 18 inches total for conventional airplanes) and in doing so, should not exceed the design load factors for the landing conditions.

In designing shock absorbers it is usually necessary to consider the following:

1. Gross weight of airplane.
2. Static load on shock absorber.
3. Design load factors.
4. Design load for the principal landing conditions.
5. Required height of free drop in inches equal to 0.36 (stalling speed in m.p.h.) but not greater than 18 inches.
6. Weight of chassis.
7. Vertical movement of wheel center.
8. Maximum tire deflection.
9. Tire size.
10. Tire inflation.
11. Shock absorber strut travel desired.
12. Required length of shock absorber strut under static load in level-landing and three-point landing position.
13. Minimum or maximum length, or both, required for fully compressed condition.
14. Minimum or maximum length, or both, required for fully extended condition.

It is often desirable to draw up a three-view line drawing of the landing gear to show the configuration of the various members.

It is not generally desirable to place the shock absorbers at too great an angle to the vertical for the best results. A limiting angle of 45 degrees may be assumed although not more than half that would be more desirable.

Retraction of Landing Gear

No high speed airplane should have a non-retracting gear if the best possible performance is desired.

The simplest form of retraction is one that employs only one motion—either in a plane parallel to the line of flight or in a plane perpendicular to the line of flight.

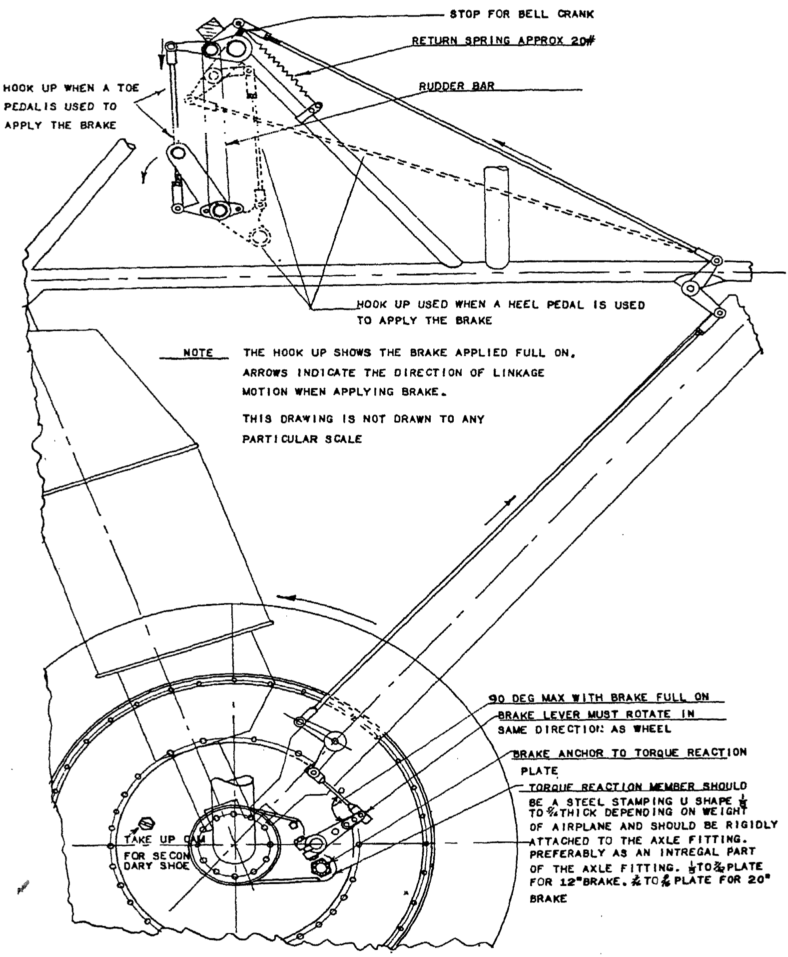


FIGURE 76

For moderately small airplanes of the single engine type, retraction sideways is best since sufficient room is available in the root of the wing, whereas retraction rearward will still leave at least half of the wheel exposed.

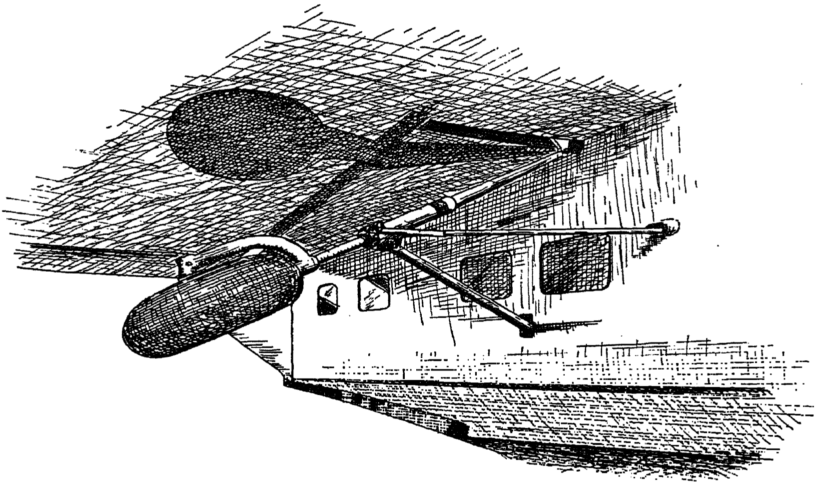


FIGURE 77. THE RETRACTABLE LANDING GEAR USED ON THE FAIRCHILD AMPHIBION

This suggests a possible solution for retracting the landing gear into a high wing.

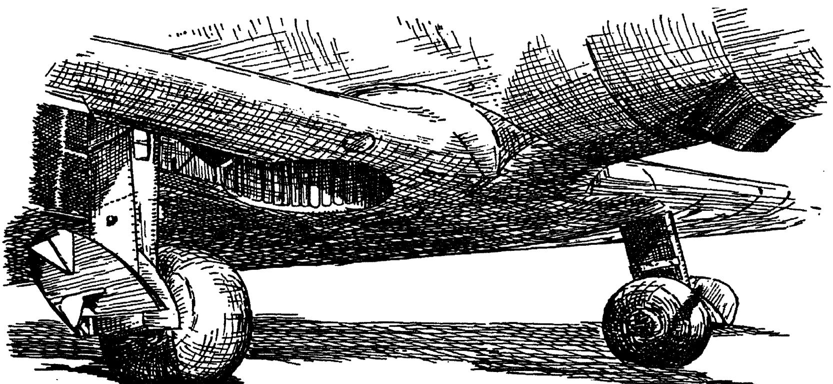


FIGURE 78. A TYPE OF RETRACTABLE LANDING GEAR USED ON AN EARLY VULTEE DESIGN

The simplest design for retraction, since the wheel is moved inwardly, recesses into the wing, and is covered by a special cover to reduce aerodynamic resistance. The wing structure may have to be modified to permit such retraction in most

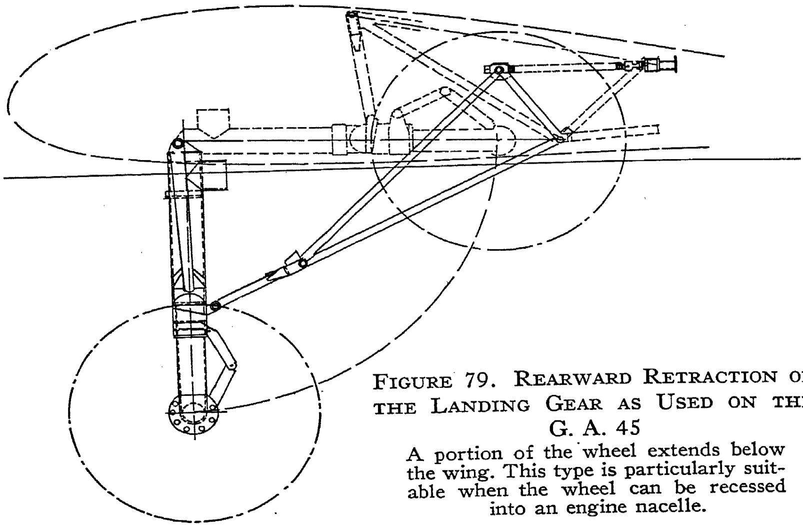


FIGURE 79. REARWARD RETRACTION OF THE LANDING GEAR AS USED ON THE G. A. 45

A portion of the wheel extends below the wing. This type is particularly suitable when the wheel can be recessed into an engine nacelle.

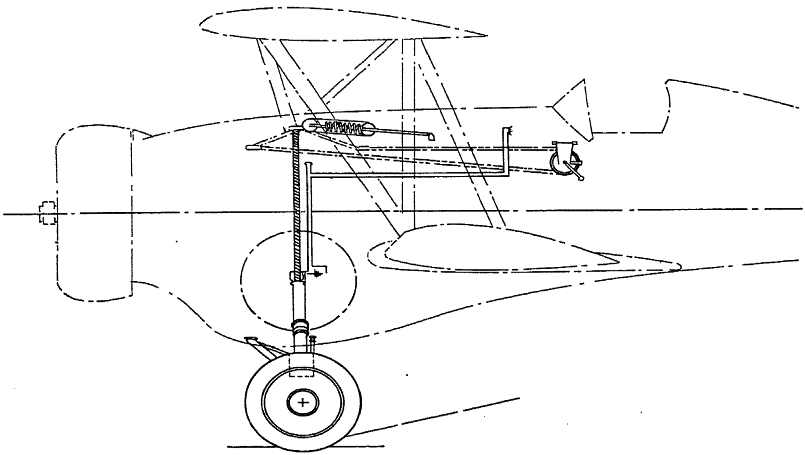


FIGURE 80. A LANDING GEAR WHICH RETRACTS INTO THE SIDE OF THE FUSELAGE

This type is used on Grumman designs, and the Curtiss Hawk.

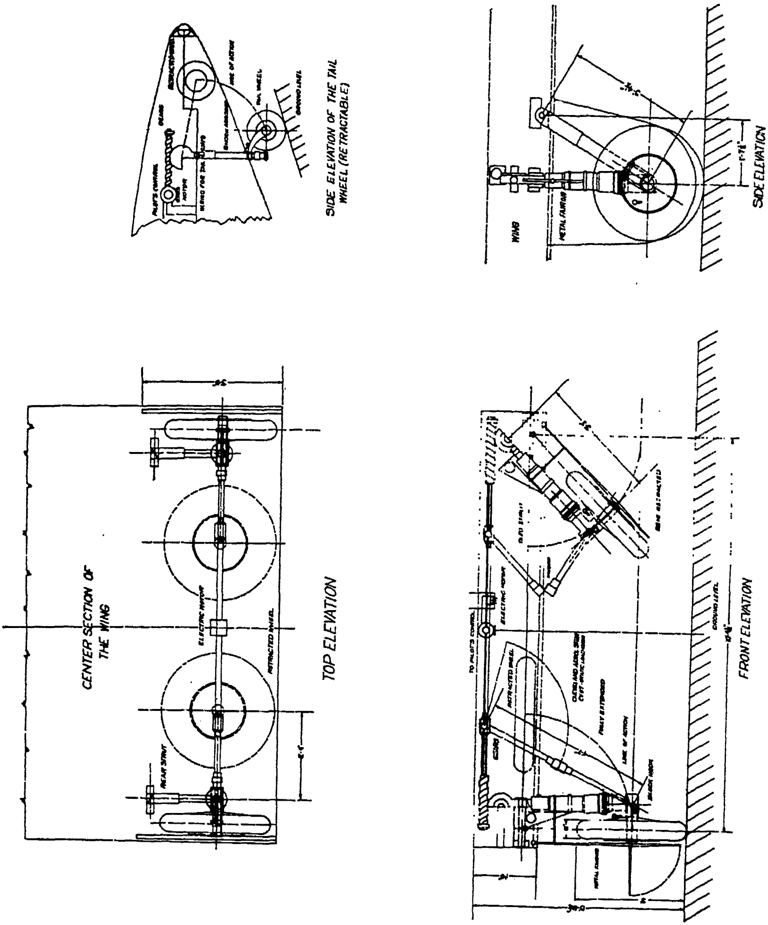


FIGURE 81

The student should study the schematic sketches of retractable landing gear here presented in order to get a good conception of the various methods that may be employed for retraction.

XIII. Tail Surfaces

The tail surfaces serve two functions. The fixed, together with the movable, portions of the tail surfaces provide stability while the movable portion in conjunction with the fixed portion provides a means for control.

It is very important that these tail surfaces be so located that they are not blanketed by the fuselage. If the fuselage has a relatively large cross section for the greater part of its length and then tapers suddenly near the tail post it is very likely that the horizontal tail surfaces will be blanketed unless the aspect ratio of these surfaces is high.

The vertical tail surfaces are most likely to be blanketed, not only by the fuselage, but by the horizontal tail surfaces as well, especially when the airplane is at a high angle of attack. In order to minimize this effect, it would be desirable to get some of the vertical tail surfaces below the horizontal tail surfaces. This can be done easily if the airplane employs the tri-cycle landing gear, but if it employs the conventional type of landing gear, it is practically impossible to get more than a very little below the longitudinal axis of the fuselage.

Aerodynamically, it would be nice to have a large aspect ratio for both the vertical and horizontal tail surfaces, but unfortunately the greater the aspect ratio the more difficult it becomes to get an efficient structure that will be rigid. Since the movable surfaces are a reasonably large proportion of the total area, the fixed portion, which supports the movable surfaces, must contain all the necessary structure. If the aspect ratio is too great for the area, there is relatively little depth with the result that, under a load, the fixed surfaces may deflect so much that the hinges of the movable surfaces bind.

The proportion of the movable surfaces to the whole depends upon the degree of control desired. A large movable surface, for example, needs less angular deflection than a smaller-sized surface. If the airplane is to be very maneuverable, it is desirable to have relatively large surfaces. In any case it is necessary to have sufficient control at the slowest and at the highest speeds the airplane will attain.

The elevator should be able to trim the airplane at the lowest, or stall, speed, at which time the elevator will usually have its maximum angular deflection upward. The elevator should also be able to trim the airplane practically at zero lift, at which the elevator may have its maximum angular deflection downward.

Control, however, is not measured alone by the change in angle of trim of the airplane caused by a definite angular deflection of the elevator (and the discussion here applies equally to the vertical tail surfaces) but also by the hinge moments produced. If for the same angular deflection of the movable surface one has greater hinge moments than another, it should be obvious that the one with the smaller hinge moments can be actuated far more quickly, and the response of the entire airplane will be quicker also.

The magnitude of these hinge moments is becoming an increasingly important problem. There are several solutions available. In some cases, as the aileron, for example, a smaller chord so that the ratio of the chord of the movable surface to that of the entire surface is 15 to 20 per cent helps materially in reducing the hinge moments. This necessitates a larger span in order to get the same total control but unfortunately the rudder or elevator seldom has the ratio of its chord to the chord of the complete surface less than 45 or 50 per cent. In order to reduce the hinge moments, the surfaces may be partially aerodynamically balanced either by having the hinge line of the movable surface somewhat in rear of its leading edge; or by having a small movable surface or tab attached near the trailing edge of the main movable surface. This small surface has an angular deflection opposite to that required for the main movable surface.

If the tab is small or its setting fixed and changed only when the load conditions change the center of gravity, then its purpose is for trim only, and is known as a "trimming tab." It takes the place of the adjustable stabilizer.

If the tab can be controlled from the cockpit, it may be used to operate the larger surface and is then called a control-tab or a servo-tab.

Aerodynamic balance is generally used, even if trailing edge tabs are present. The design of this balance is very critical and is still the subject of much experimental work. For greater effectiveness, a slot in front of the leading edge of the balance is provided. Although this slot helps to increase the effectiveness of the movable surfaces, yet, more often than not, the relatively large gap caused by the slot increases the parasite drag.

Great care should be taken in designing the leading edge of the aerodynamic balance so that it is not too sharp and does not project too far above the upper or lower contour of the fixed surface when the movable surface is deflected. Such projections collect ice very quickly under icing conditions and may lead to unbalance of the control surface, or jamming of the controls.

Design details of the tail surfaces are given under the various paragraph heads immediately following.

Airfoil Sections

Symmetrical airfoils are usually used for tail surfaces so that equal effectiveness per degree deflection may be obtained for both up and down movements.

TABLE 28

Station in Per Cent of Chord	*N.A.C.A. 0006* *Upper + Lower —*	*N.A.C.A. 0009* *Upper + Lower —*	*N.A.C.A. 0012* *Upper + Lower —*
0	0	0	0
1.25	± .947	±1.420	±1.894
2.50	±1.307	±1.961	±2.615
5.00	±1.777	±2.666	±3.555
7.5	±2.100	±3.150	±4.200
10.	±2.341	±3.512	±4.683
15.	±2.673	±4.009	±5.345
20.	±2.869	±4.303	±5.738
25.	±2.971	±4.456	±5.941
30.	±3.001	±4.501	±6.002
40.	±2.902	±4.352	±5.803
50.	±2.647	±3.971	±5.294
60.	±2.282	±3.423	±4.563
70.	±1.832	±2.748	±3.664
80.	±1.312	±1.967	±2.623
90.	± .724	±1.086	±1.448
95.	± .403	±0.605	± .807
100.	±(.063)	± .095	±(.126)
100.	0	0	0

The airfoil section used should have a thickness ratio of at least 8 or 9 per cent and not more than 12 per cent. Unless the tail surfaces are exceptionally large the same airfoil is used from tip to root without taper in thickness ratio.

The N.A.C.A. 0009 and 0012 series are recommended. The ordinates of these airfoils are given in Table 28.

Aerodynamic Balance

Different types of aerodynamic balance for movable surfaces are illustrated for ailerons in Chapter XI. The same type of aerodynamic balance may be used for the elevator and the rudder.

Flutter Prevention

The Department of Commerce has suggested the following general principles to be observed for flutter prevention on all airplanes in the design of control surfaces and control systems.

1. Structural stiffness.
2. Elimination of all play in hinges and control system joints.
3. Rigid interconnections between ailerons and between elevators.
4. A relatively low amount of aerodynamic balance.
5. High frictional damping.
6. Adequate wing fillets and fairing.
7. Sharp leading edges on movable surfaces should be avoided.

The following features are considered essential for flutter prevention.

8. When separate elevators are used, the interconnecting structure which is required must be as stiff in torsion as practicable.
9. Trailing edge controls should be irreversible, relatively rigid, and the tab installation should be designed as to prevent any development of free motion of the tab. When the tabs are completely statically balanced, irreversible controls are not required provided that the tab control system cannot be manipulated abruptly through a large range. A small amount of overbalance of the tab obtained by putting its center of gravity ahead of the hinge line will help to offset harmful effects of any possible looseness or tab control system flexibility.
10. When trailing edge tabs are used to assist in moving the main surface, the areas and relative movements must be so proportioned that the main surface is not aerodynamically overbalanced at any time.

11. Experimental determination of the natural frequency of vibration of certain components of the airplane may be desirable in cases where dangerously low frequencies or the coincidence of the natural frequencies of two or more structural components exist. Where such cases do occur, re-design is necessary to change such natural frequencies.
12. For airplanes with a speed greater than 150 miles an hour, it is desirable to have the rudder and elevator dynamically balanced. The criterion for such balancing is a dynamic balance coefficient of not more than 0.08 calculated in the following manner with reference to Figure 82.

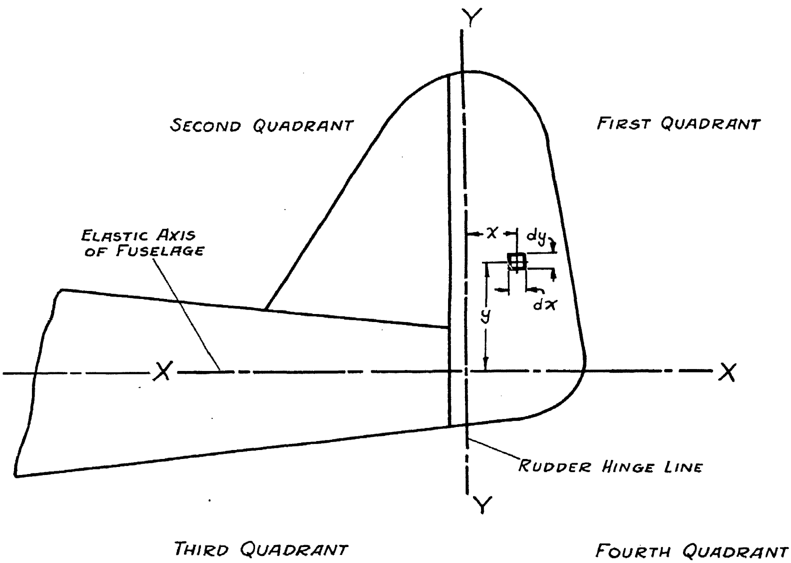


FIGURE 82

The dynamic balance coefficient is non-dimensional, which may be expressed by the following formula:

$$\left[\iint \rho \, xydxdy\right] \div \left[S\int \rho \, dxdy \right] \leqq 0.08$$

The numerator is the resultant product of inertia of the control surface, and the denominator is the product of the mass and the aerodynamic area of the control surface. In the formula expressed

above ρ is the unit weight or mass, x is the distance between the center of gravity of the mass and the X-axis, y is the distance between the center of gravity of the mass and the Y-axis, and dx·dy is the area of the mass under consideration, $S = \Sigma\, dxdy$.

The products of inertia in the first and fourth quadrants are considered positive; the products of inertia in the second and third quadrants are considered negative.

The calculations required are illustrated by the simple problem given below with reference to Figure 83 which may be assumed to

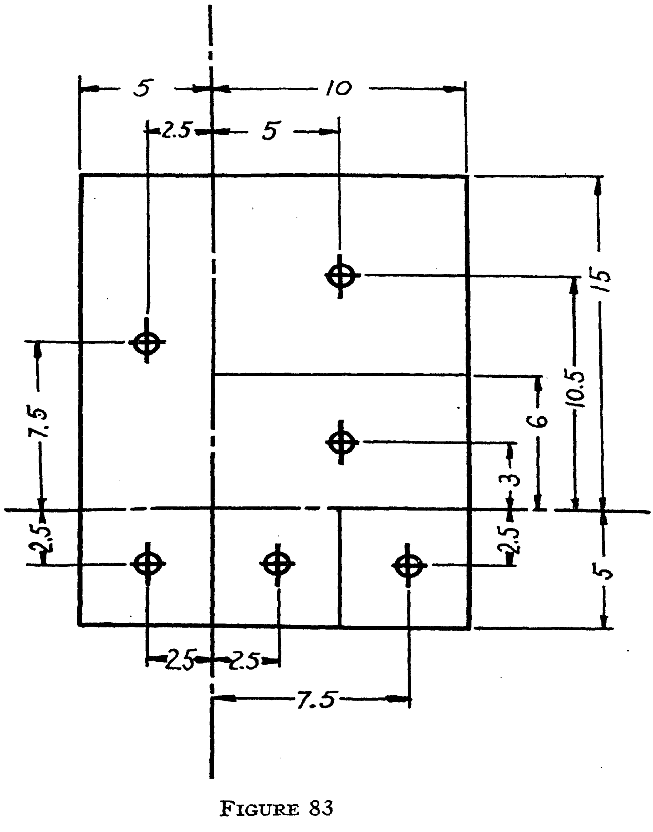


FIGURE 83

be a rudder with the X-X axis the torsional axis of the fuselage, and the Y-Y axis the hinge line of the balanced rudder. The calculations are presented in tabular form, and are self-explanatory.

TABLE 29

Quadrant	ρ Lb. per Sq. In.	X In.	Y In.	$dx\,dy$ Sq. In.	$\rho xy\, axdy$	$\rho\, dx\, dy$
1	0.010	5.0	10.5	90	47.25	.90
1	0.015	5.0	3.0	60	13.50	.90
2	0.010	2.5	7.5	75	—14.06	.75
3	0.010	2.5	2.5	25	— 1.56	.25
4	0.015	2.5	2.5	25	2.34	.375
4	0.010	7.5	2.5	25	4.69	.25
				280	51.16	3.425

The dynamic balance coefficient for the rudder would be

$$C_{db} = \frac{51.16}{280\ (3.425)} = .053+$$

It is desirable to choose very small areas in making these calculations in order to get as much accuracy as possible. A rudder, for example, having a structure with fabric covering, and static balance, will have a greater mass along the hinge line or near the leading edge than closer to the trailing edge. The example cited above is merely to illustrate the method used in the calculations.

A dynamically unbalanced surface will always have a positive resultant product of inertia.

HORIZONTAL TAIL SURFACES

1. *Location*

The horizontal tail surfaces should be so located that any blanketing by the wing or the fuselage is avoided. Partial blanketing usually exists, however, but certain features may be incorporated to limit the effect of blanketing.

For ease in assembly and disassembly the horizontal tail surfaces are attached to the top of the fuselage, especially if tubular steel construction is used for both the tail surfaces and the fuselage. When sheet metal construction is used, the horizontal tail surfaces may be located nearer the longitudinal center line of the rear portion of the fuselage and still obtain the necessary rigidity.

In some cases, the location of the horizontal tail surfaces is determined by clearance requirements for the elevator, as shown in Figure 84, when the elevator is deflected downward through its total angular range, and with the tail wheel assembly fully deflected.

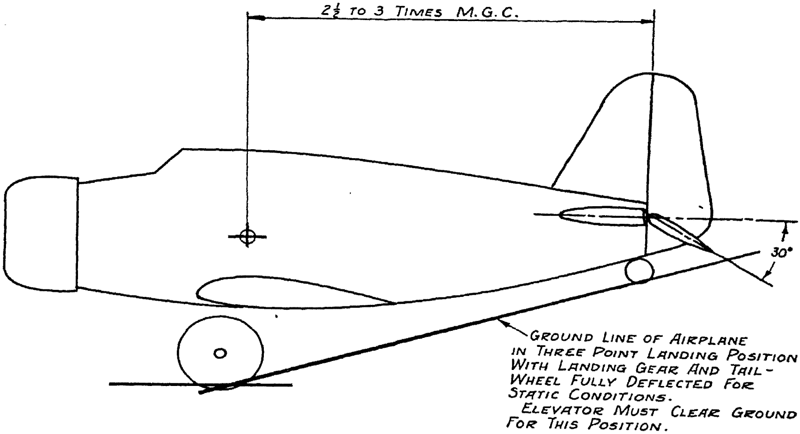


FIGURE 84

The effect of various locations of the horizontal tail surfaces with respect to the fuselage may be studied by consulting the illustrations accompanying Chapter XV.

Conventional airplanes of today locate the tail surfaces about 2½ to 3 chord lengths (mean geometric chord of the wing) behind the center of gravity, so that the observance of this rule will assure reasonable static longitudinal stability.

2. *Movement*

Elevators are designed to have an equal angular movement up and down from neutral. About 30-degree movement is considered maximum and with efficient design a 25-degree deflection up and a 25-degree deflection down should be sufficient.

The stabilizer may be adjusted through a small angular displacement either on the ground or in the air from the cockpit—usually the latter, if at all, since trimming tabs are displacing adjustable stabilizers.

If an adjustable stabilizer is used, a total of 6- to 8-degree movement (about 5 degrees up and 3 degrees down) is usually used.

3. *Aspect Ratio*

The aspect ratio of the tail surfaces should be as high as possible in order to avoid blanketing of the structure to which they are attached. Aspect ratios greater than 6 are seldom used unless they can be braced adequately.

In proportioning the tail surfaces, it is not desirable to start with the aspect ratio because the fuselage section increases the span of the tail surfaces seemingly beyond the desirable limit.

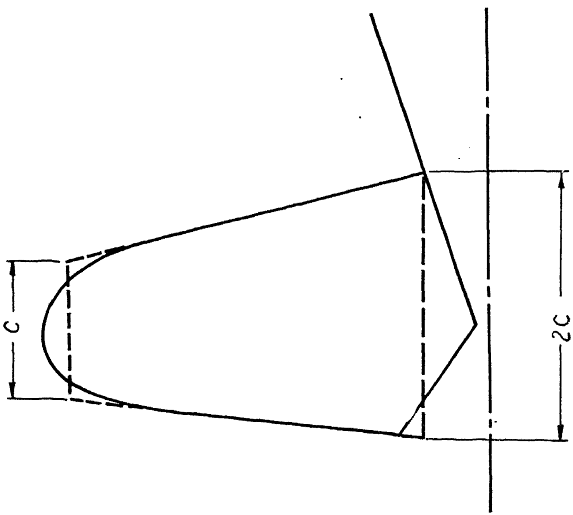


FIGURE 85. PROPORTIONING THE HORIZONTAL TAIL SURFACES

Figure 85 shows a reasonable method for proportioning the tail surfaces by considering only half of the horizontal tail surfaces at a time. By making the span about $1\frac{1}{2}$ times the mean chord, a reasonable aspect ratio for the tail surfaces is obtained.

For correcting airfoil data from the given aspect ratio to that of the tail surfaces, the aspect ratio is calculated on the basis of the square of the span length from tip to tip divided by the area including that covered by the fuselage. In other words, exactly the same procedure is followed as in calculating the aspect ratio of the wing.

The position of the horizontal tail surfaces will also affect the aerodynamic characteristics and a few cases are shown in the illustrations of horizontal tail surfaces shown in Chapter XV.

4. *Incidence*

The incidence of the horizontal tail surfaces is determined by the amount of downwash from the wing, its relative location with respect to the wing, and the moment required to obtain the required trim angle. These considerations are discussed extensively in Chapter XV.

On small airplanes it has been customary to make the stabilizer adjustable through a limited angular range—about 3 degrees up and 3 degrees down. This adjustment has been possible either on the ground or in the air by means of a control located in the pilot's cockpit. The adjustment in the air is preferable. On the large transport airplanes, variations in trim—the object of the adjustable stabilizer—are obtained by means of trailing edge tabs.

5. *Dihedral*

Normally horizontal tail surfaces are not given any dihedral, but it has been found that the effectiveness of the horizontal tail surfaces can be increased considerably, particularly at high angles of attack, by incorporating some dihedral in the horizontal tail surfaces. How large the dihedral angle should be has not been definitely established so that for purposes of symmetry the span line of the tail surfaces is made parallel to the span line of the wings.

6. *Area*

Examination of airplanes of all sizes reveals that the ratio of the horizontal tail surfaces to the effective wing area varies from 15 to 20 per cent. The greater the tail length is, in terms of the wing chord, the smaller percentage area is required. Wings equipped with lift increase device usually require that the percentage area of the horizontal tail surfaces be greater than if the wings were not so equipped.

The elevator area varies from 35 to 45 per cent of the horizontal tail surface area.

The shaded portion of Figure 86 is considered the effective tail surfaces. That portion covered or blanketed by the fuselage is not considered effective.

The elevator area is the entire area measured up to the leading edge represented by the solid line in Figure 86.

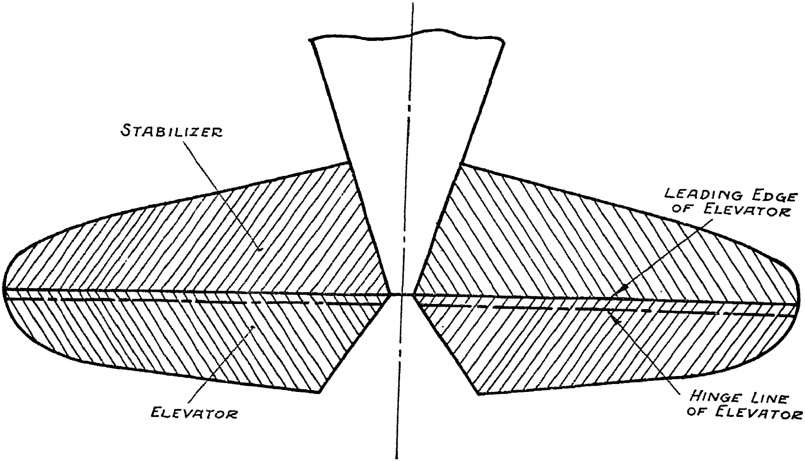


FIGURE 86

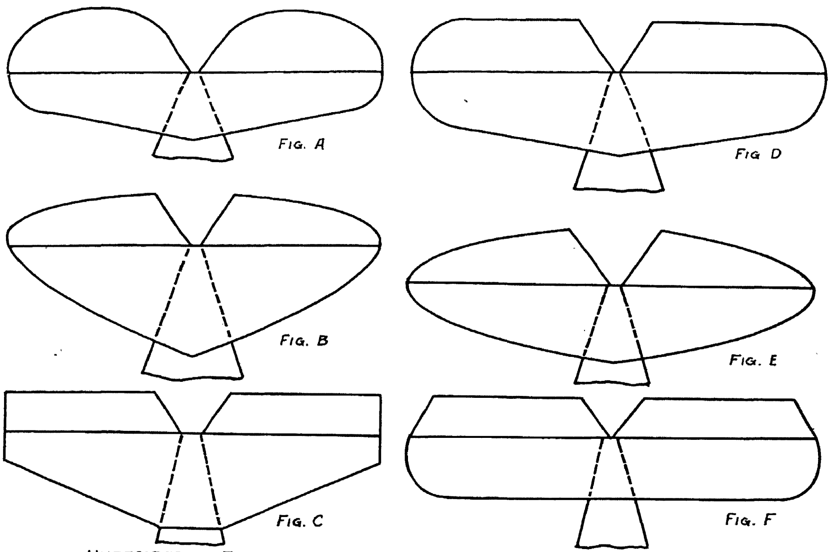


FIGURE 87. POSSIBLE PLANFORMS FOR HORIZONTAL TAIL SURFACES

7. *Planforms*

Figures 87 A, B and C show several generally undesirable planforms for horizontal tail surfaces. Figure 87 A shows an elevator planform which is difficult to construct and has nothing particularly in its favor except its appearance.

Figure B shows the type of planform which has too much area near the fuselage where it may become blanketed and lose much of its effectiveness. If these tail surfaces were located on a narrow fuselage or where the airflow is reasonably unobstructed, the planform shown would be satisfactory.

Figure C shows a type of planform with the same drawbacks as Figure B. The alternative dotted planform is undesirable since the leading edge of the elevator will be determined by the depth of the elevator spar so that the maximum thickness is just at the leading edge; the result is a poor airfoil and a poor elevator.

Figures D, E and F are well-proportioned planforms recommended for horizontal tail surfaces.

VERTICAL TAIL SURFACES

1. *Location*

The vertical tail surfaces are, almost without exception, located above the horizontal tail surfaces, in order to centralize control systems and simplify the supporting structure contained in the fuselage.

It is desirable to locate about half of the rudder below the axis of symmetry of the fuselage but this may not be possible because of required clearance with the ground.

Figure 88 shows some likely arrangements and planforms for the vertical tail surfaces.

2. *Planform*

The discussion under HORIZONTAL TAIL SURFACES applies here, although the vertical tail surfaces may be slightly broader at the base in order to assure rigid construction.

3. *Aspect Ratio*

The aspect ratio of the vertical tail surfaces should be between 2 and 4. It is difficult to state exactly what the aspect ratio of the vertical tail surfaces may be. Figure 89 indicates what may be assumed for the effective area and effective span of the vertical tail surfaces, in order to calculate an aspect ratio from the formula (Effective Span)^2 $\div$ Effective Area.

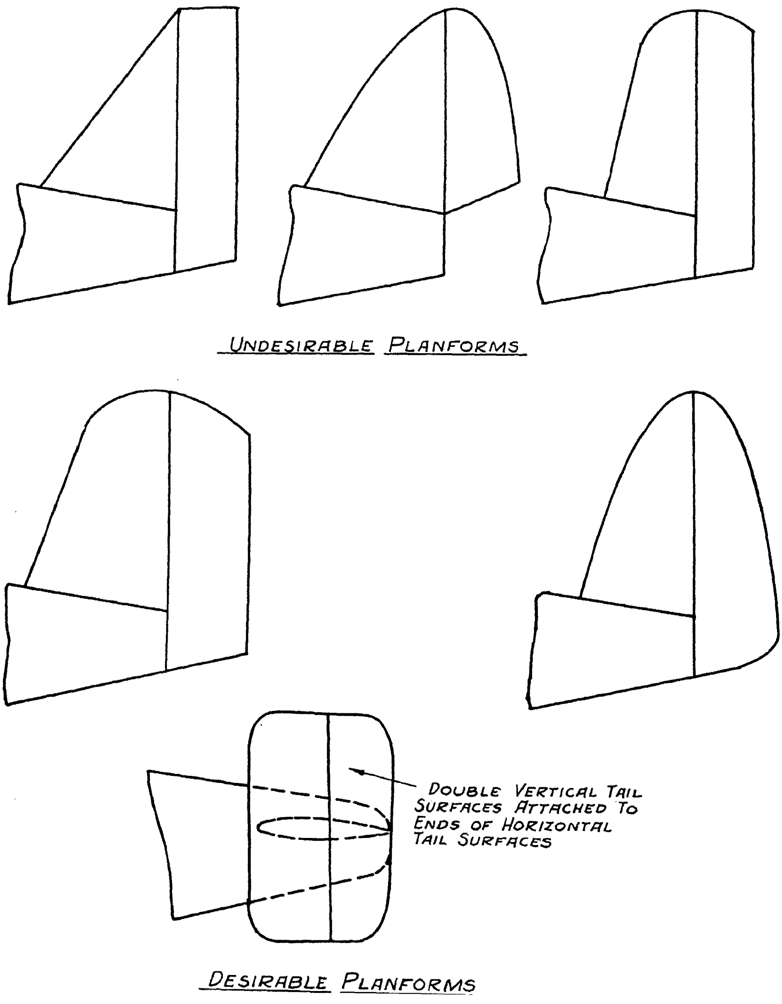


FIGURE 88. POSSIBLE PLANFORMS FOR VERTICAL TAIL SURFACES

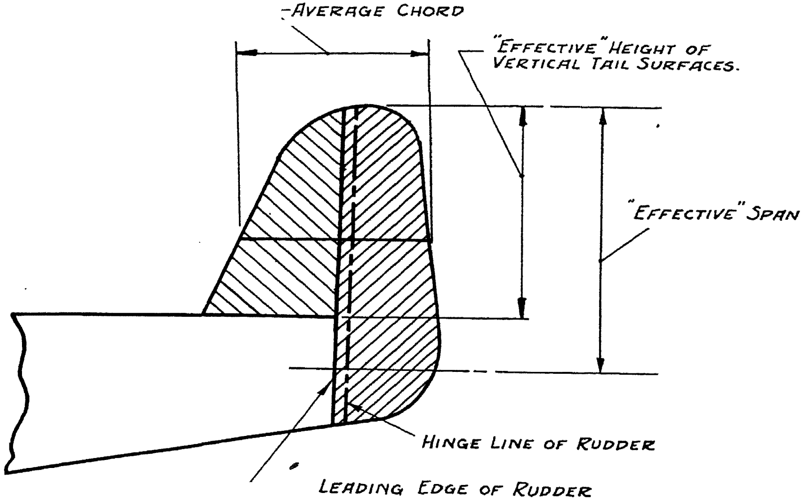


FIGURE 89

In proportioning the vertical tail surfaces refer to Figure 89. The "effective" height should be from $1\frac{1}{2}$ to twice the average chord.

Trailing Edge Tabs

Trailing edge tabs on movable surfaces are popular at the present time. These consist of an inset adjustable portion as shown in Figure 90 where a tab is applied to the rudder.

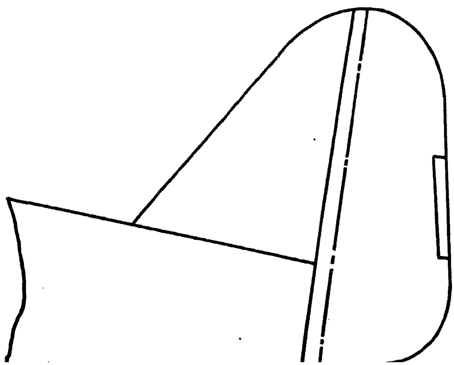


FIGURE 90. TRAILING EDGE TAB ON THE RUDDER

A tab on the aileron is used to overcome engine torque; it may be used with the rudder for the same purpose as the offset fin; and on the elevator for the same purpose as the adjustable stabilizer.

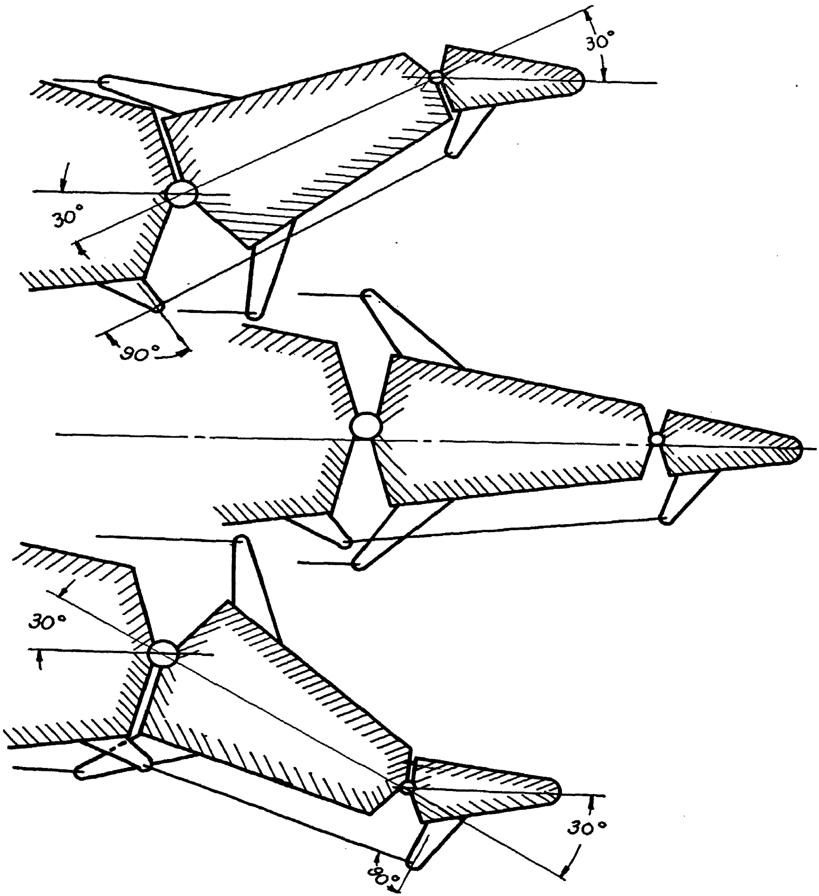


FIGURE 91. BALANCING TABS ON TRAILING EDGE OF MOVABLE SURFACES

These tabs may be used either as trimming tabs—which may be adjusted on the ground or in the air—or they may be used as servo-control tabs. Balancing tabs are illustrated in Figure 91, and a servo-control tab is illustrated in Figure 92.

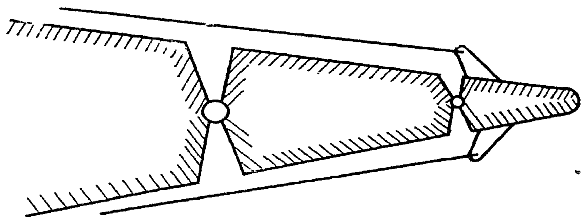


FIGURE 92. THE SERVO-CONTROL TAB APPLIED TO A MOVABLE SURFACE

Trimming tabs have a chord varying from 5 to 10 per cent of the movable surface chord and approaching 25 per cent of the chord if used as servo-control tabs. The aspect ratio should be as high as possible—varying usually from as low as 5 to high as 20.

XIV. The Fuselage

Even though comparatively little thought seems to have been given to the size, shape and structure of the fuselage up to now, it has been practically designed in most of its major elements while the pilot's cockpit, the passenger cabin, the placement of the wing and the balance diagram were under consideration.

If the fuselage is to be designed for a single engine airplane, the type of engine and its cowling determine the nose treatment of the fuselage. The pilot's cockpit governs the development of the windshield and that section of the fuselage directly behind the engine. If there is no nose engine, as in multi-engine designs, the pilot's cockpit has even a greater influence on the development of the front section of the fuselage.

The main section of the fuselage is built around the passenger cabin, and then the rear portion of the fuselage gradually tapers back to the tail post.

Since all these items affect the size and shape of the fuselage, it is obvious that the arrangement and general dimensions of the pilot's cockpit and the passenger cabin are the primary considerations. Only after all this is done should the contours of the fuselage be settled. The cross sections of the fuselage will vary from circular at the nose to slightly elliptical at the pilot's cockpit, and then to an elliptical section somewhat modified by the intersection of the root of the wing, and then to an elliptical or oval section, sometimes even rectangular (with the long side vertical) at the tail post. The object of approaching the elliptical section is that it represents the best compromise between the square or rectangular section which is most efficient for space utilization (as cabins, cockpits, etc.) and the circular section which is most efficient from a structural point of view for reinforced monocoque structure, and for aerodynamic reasons.

The longitudinal cross sections, both vertical and horizontal, should approach airfoil or airship forms in order to reduce the aerodynamic resistance to a minimum. It is not an easy task to blend the vertical cross sections from nose to tail, and to obtain reasonable longitudinal cross sections. To do so on paper requires the fairing or blending of curves by means of projective geometry. It is usually easier and quicker to build a mock-up in which each part

of the airplane is represented to full-size scale, except for only minor modifications, by means of wood and cardboard. The wooden members usually represent the larger tubular or fabricated members while the flat cardboard, or in some cases plywood, represent the flat metal sheets.

By building a mock-up, a fuselage form may be evolved which may use flat sheet material without resorting to double curvatures and so help in materially reducing the production costs. If the covering has a curvature in one direction only, a flat sheet may be used. If, however, there is a curvature at an angle to the main radius of curvature, the covering has what is known as a double curvature which must be produced by forming the sheet either by gradual "bumping" or by hydraulic presses over special dies.

The mock-up also helps to indicate whether any other modifications in arrangements are desirable. This method is a simple and inexpensive way to solve many problems usually too intricate for the drafting board, and it also helps to visualize a design far better than a series of drawings could possibly do.

Assuming, then, that the outside contour has been definitely established, the next step is to locate the structure. Wherever concentrated loads have to be transmitted to the fuselage—as from the wing spars, or landing gear members, or tail surface spars—a double or reinforced frame has to be located. For example, there would be double frames at the front and rear spar locations of the wing. Between these double frames would be located single frames spaced from 10 to 24 inches apart. The spacing of these frames may be governed by the window locations. How often do the front spar frames intersect a window which has been so painstakingly placed there to give the passenger the proper vision! It is generally not desirable to break or distort these main frames just to accommodate a window. In such a case the difficulty may be adjusted by relocating the window, or relocating the spar so that the frames may be changed.

Naturally there will be frames at each side of the door for such frames will provide convenient hinge supports as well as door frame supports. The front and rear cabin wall will determine another station where frames should be located to provide anchorage for the cabin. Such considerations gradually "build-up" the structure until a few intermediate frames are added just to cut down the unsupported length between frames. Similar considerations determine the location of the longitudinal members.

The fuselage structure is determined gradually, and often, when it is not possible to compromise, radical changes in the wing plan form or in the interior arrangement may result. For example, it may be desirable to have the front spar double frame intersect the fuselage at the front cabin wall, but in order to do so, the wing has to be given an appreciable sweep back in order to satisfy not only the particular condition just outlined but also to obtain the proper location of the mean geometric chord with respect to the center of gravity.

The various factors affecting the fuselage design are recapitulated below.

Shape of Fuselage

The length of the fuselage is determined by the cockpit and cabin considerations in the front and the location of the tail surfaces and the tail wheel at the rear; the overall depth and width by cabin requirements. For efficient structural design, a circular cross section is desired, but this shape is inefficient for accommodating a rectangular cross section required by the cabin so that the natural compromise is an oval, perhaps with smaller width at the top than at the bottom.

The various cross sections of the fuselage may vary from a circular section at the engine mount to an oval section again and finally a circle or a rectangle at the tail post. It is therefore very important to fair one section into another very carefully, and to avoid double curvatures (two curvatures at an angle to each other) wherever possible.

In order to do this simply, a mock-up made up of wood frames with cardboard or plywood covering is usually made in the shop. If the flat sheets of cardboard or plywood have contact with the entire periphery of the frames which the sheet covers, then no double curvature exists. This is a trial and error method, but usually quicker, easier and surer than working out the problem by descriptive geometry methods. Small double curvatures may be obtained without too much trouble by slight pounding or bumping of the sheet before it is applied to the frames.

Location of Frames

The frames, whether circular rings or reinforced frames, are located from 10 to 24 inches apart with the closer spacing near the cabin and cockpit and gradually increasing towards the tail post.

The frames may rest on the longitudinal members or on the outside covering with cut-outs for the longitudinal members.

Reinforced frames are located wherever concentrated loads are applied as at attachment of wing spars or landing gear members. In order to reduce structural weight, it is decidedly advisable to locate several external attachment points on the same frame wherever possible.

In general the locating of frames will be determined first by the location of spars, landing gear, doors and windows. Reinforced frames are required at every one of these locations of external attachments in order to distribute the loads into the metal covering evenly. After these reinforced frames have been placed, the intermediate frames can be placed at proper intervals.

Reinforced frames may be similar to intermediate frames in construction except for the thicker material or local reinforcements; or they may be two frames, back to back, but spaced a few inches apart, with tying members between them to obtain rigidity and continuity of structure.

Location of Longitudinal Members

Longitudinal members are spaced from 6 to 12 inches apart around the periphery of the largest cross section. Since the cross sections gradually diminish in size, the spacing is closer towards the tail post so that alternate members may be stopped at a forward frame. It is desirable not to end all longitudinal members at the same frame.

To assure a continuous structure, it is customary to make the longitudinal members continuous rather than breaking them at every frame.

Sections for Structural Members

Extruded sections may be used where the curvature is not severe such as longitudinal members, or stringers, used in fuselages or wings.

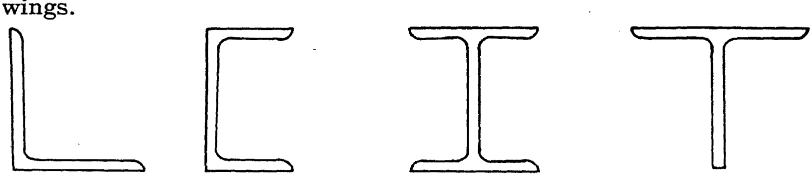


FIGURE 94. TYPICAL EXTRUDED ALUMINUM ALLOY SECTIONS

Formed sections resemble extruded sections but are made of sheet metal. These sections may be curved and are used for making ribs, intermediate frames for fuselages and hulls, and wherever lighter members are desired and where extruded members would be hard to bend to the desired shape.

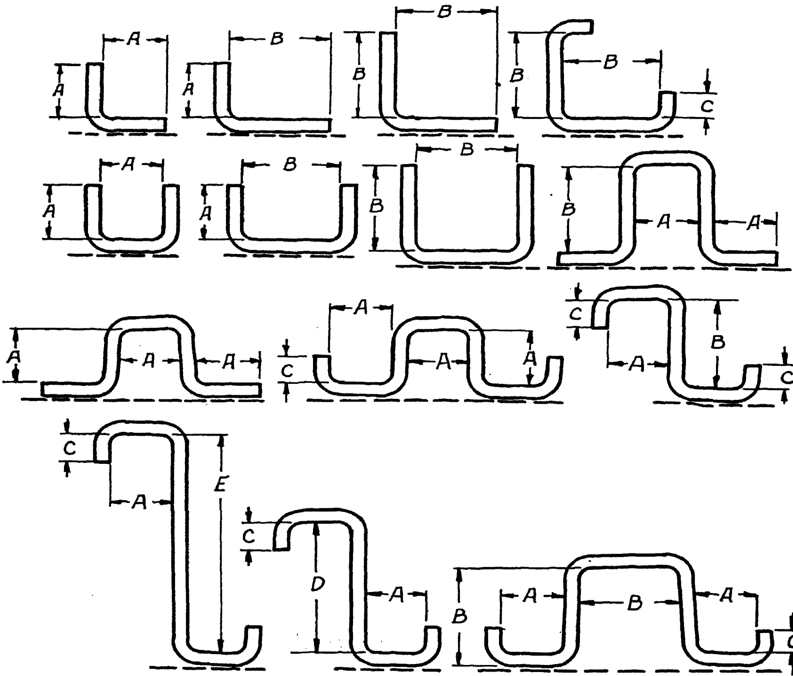


FIGURE 95. FORMED METAL SECTIONS
Refer to Table 30 for representative dimensions.

Floor

The floor of the cabin rests on the frames either directly or through an intermediate structure—usually the latter—in order to permit application of soundproofing materials and occasional replacement if desired.

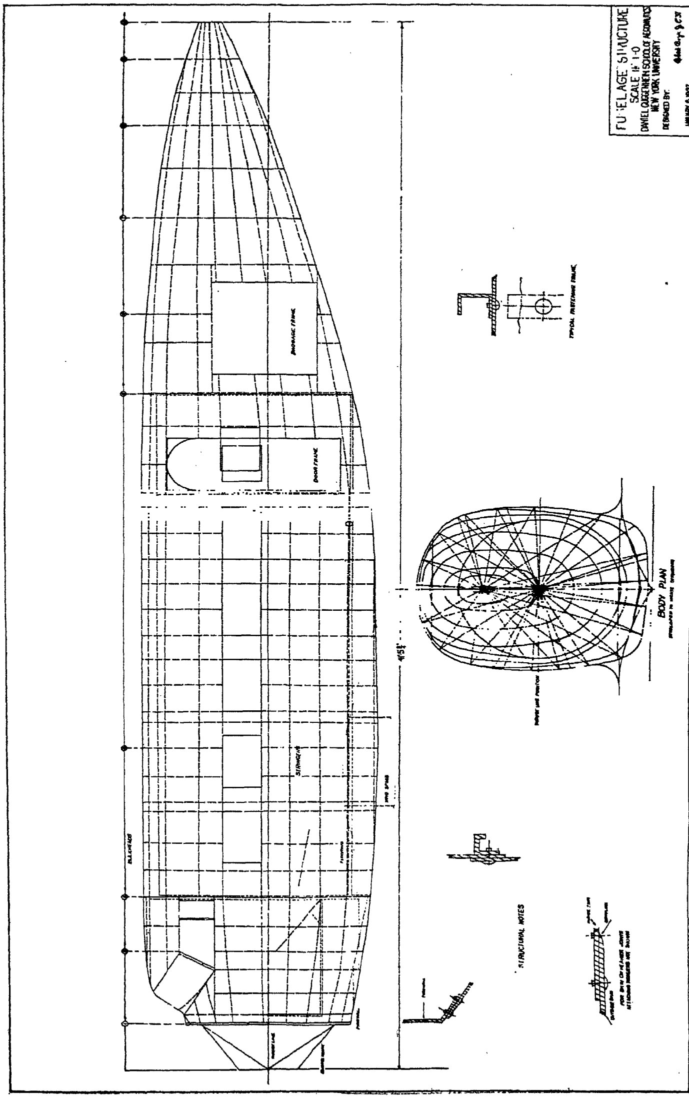


FIGURE 96
(The original drawing, to which the scale applies, was 34⅝" x 64½".)

TABLE 30

B & S No.	*Gauge t*	*A*	*B*	*C*	*D*	*E*	*E*	*E*
28-21	.012-.028	3/8	5/8	3/16	1	1 1/4	1 1/2	1 3/4
20-18	.032-.040	1/2	7/8	1/4	1 1/4	1 1/2	1 3/4	2
17-16	.045-.051	5/8	1	5/16	1 1/2	2	2 1/2	3
15-14	.057-.064	3/4	1 1/4	3/8	2	2 1/2	3	3 1/2
13-12	.072-.081	1	1 5/8	7/16	2 1/2	3	3 1/2	4
11-10	.091-.102	1 1/4	2	1/2	3	3 1/2	4	5

Baggage Compartments

Baggage compartments and the like are lined with a secondary structure of metal sheet, either flat or corrugated (may be covered with canvas or felt for soundproofing purposes). This secondary structure is attached to the frames or longitudinal members at appropriate points.

Access Doors

Where access from the outside is required, readily removable access doors which usually employ some form of cowl fastener for quick opening should be provided in the fuselage design.

Batteries should be removable from the outside and not from the inside of the cabin or cockpit; likewise the water tank should be fillable from the outside. Since these two items are not removable and access is not required at every landing, the access doors do not have to be as easily opened as the doors to the baggage compartments.

XV. Longitudinal Stability

Imagine an airplane flying along in horizontal flight at constant speed struck by a sudden gust of wind, either on the wing or on the tail surfaces, so that the airplane would be forced suddenly nose upward or nose downward. Without any control exerted by the pilot, or any change made in the throttle setting, the airplane should return after the gust to its angle of attack before being struck by the gust and continue to fly on an even keel in horizontal flight. It may not return to its original angle of attack instantaneously (as a matter of fact it is undesirable for an airplane to do so because such action may make it decidedly uncomfortable for the passengers) but the airplane may oscillate back and forth with the oscillations gradually diminishing.

The property which tends to return an airplane to its original condition of equilibrium or line of motion (fly at a given angle of attack at constant speed) when disturbed from that condition of equilibrium or steady motion by an external force (which may be simulated by the pilot, for example, when he depresses the elevator but returns it to its original position almost immediately) due to forces or moments developed inherently of such character as to counteract the disturbing force or moments is known as *stability*. An example of longitudinal stability has been cited in the previous paragraph. There are similar cases of stability about the longitudinal axis, or lateral stability, and about the vertical or normal axis, or directional stability.

There are two distinct types of stability, aside from their relationships to particular axes. An airplane, for example, is so designed that at one angle of attack, the resulting moment of all forces about the center of gravity will be zero. This satisfies the condition of equilibrium together with the fact that the sum of all forces along the three axes equals zero, but this is known as *static* longitudinal stability. It was mentioned earlier that the airplane does not come back to its original condition of equilibrium immediately but continues to oscillate. This oscillation is a manifestation of dynamic longitudinal stability.

There are also three different states of static stability and of dynamic stability, whether applied to longitudinal stability, lateral stability or directional stability.

For example an airplane may be neutrally stable. It may be disturbed from its original attitude, and instead of coming back to it, will continue in its new attitude.

Or the airplane may be unstable: when disturbed from its original attitude, instead of coming back to it, it will go farther and farther, not remaining at any attitude.

If the airplane does come back to its original attitude, then the airplane is said to be stable.

These three examples are cases of static stability. Similar cases of dynamic stability would be:

1. Stable when the oscillations of the airplane gradually decrease to zero after they are once started.
2. Neutral when the oscillations do not decrease or increase in amplitude when once the oscillations have been set up.
3. Unstable when the oscillations increase with time when once they have been set up.

It is a comparatively easy matter to calculate the static longitudinal stability of an airplane, but the static lateral and directional stability are seldom calculated. They usually have to be determined by careful experimentation in the wind tunnel on small-scale models. By following orthodox practices, there is seldom much trouble experienced in obtaining the proper degree of stability. Obviously there must be criteria to determine whether or not an airplane is sufficiently stable. A highly maneuverable airplane, such as an acrobatic or stunt airplane, for example, should not have as much stability as an airplane designed for straightforward flying, as are passenger transport airplanes.

Dynamic stability is seldom calculated for airplanes since there are so many factors involved which would have to be arrived at by empirical methods. The calculations are long and tedious. If the airplane is built along orthodox lines it is usually dynamically stable when it is statically stable.

An outline of a suggested method for determining the static longitudinal stability of an airplane follows.

PRELIMINARY CALCULATIONS FOR STATIC LONGITUDINAL STABILITY

Static longitudinal stability of an airplane may be determined with reasonable accuracy by calculating the pitching moments about the center of gravity of the airplane, fully loaded, when the airplane is disturbed from its required angle of trim. The usual angle

of trim considered is that with controls neutral, and at that angle of attack at which maximum speed or cruising speed occurs. The maximum and cruising speeds are determined as explained in Chapter XVI.

The moments are calculated for the airplane considered as a glider, or in the power-off condition, so that the effects of propeller slipstream and propeller thrust moments about the center of gravity are neglected.

The notations used in the equation refer to those used in Figure 97.

The chord, C, of the wing refers to its mean aerodynamic chord projected onto the plane of symmetry for convenience, and taking into account sweepback, dihedral, incidence and any other possible variation.

The aerodynamic load, F_t, on the horizontal tail surfaces is assumed to act at 20 per cent of the mean aerodynamic chord of the horizontal tail surfaces for all angles of attack. For convenience, the mean geometric chord is assumed to be the mean aerodynamic chord.

The moment about the center of gravity of the resistance due to the fuselage, landing gear, engine nacelles, etc., is generally neglected.

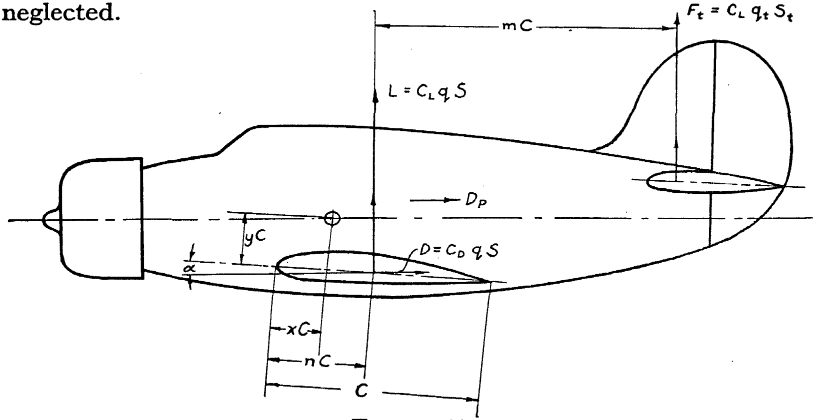


FIGURE 97

Referring to Figure 97, the moments about the center of gravity of the aerodynamic forces are:

$$M_w = -(L \cos \alpha)(n\,C - x\,C) + (L \sin \alpha)(y\,C) - (D \cos \alpha)(y\,C) - (D \sin \alpha)(n\,C - x\,C)$$

$$= (L \cos \alpha + D \sin \alpha)\ (x\,C - n\,C) + (L \sin \alpha - D \cos \alpha)\ (y\,C)$$
$$= q\,C\,s\left[(C_L \cos \alpha + C_D \sin \alpha)\ (x - n) + (C_L \sin \alpha - C_D \cos \alpha)\ (y)\right] \quad (1)$$

$$\text{or } \frac{M_w}{q\,C\,s} = \left[(C_L \cos \alpha + C_D \sin \alpha)\ (x - n) + (C_L \sin \alpha - C_D \cos \alpha)\ (y)\right] \quad . \ (1a)$$

For a high wing monoplane, with the center of gravity below the mean geometric chord, equation (1a) becomes:

$$\frac{M_w}{q\,C\,S} = \left[(C_L \cos \alpha + C_D \sin \alpha)(x - n) - (C_L \sin \alpha - C_D \cos \alpha)(y)\right] \cdot (1b)$$

The moment about the center of gravity of the air forces on the horizontal tail surfaces is:

$$M_t = -F_t\ m\ C \cos \alpha = C_{L_t}\ q_t\ S_t\ m\ C \cos \alpha\ . \quad (2)$$

Cos α is assumed equal to 1 for all practical purposes, and the horizontal component is neglected. The velocity of air over the tail surfaces is about 9/10 of the air velocity over the wing so that

$q_t = \frac{1}{2} \rho V_t^2 = .8\,q$ so that expression (2)

becomes $$M_t = -.8\,C_{L_t}\ q\ S_t\ m\ C \quad (2a)$$

or $$\frac{M_t}{q\,C\,S} = -.8\ C_{L_t}\left(\frac{S_t}{S}\right) m. \quad (2b)$$

The total moment, due to the aerodynamic forces about the center of gravity, then becomes

$$M_{C_g} = M_w + M_t \quad (3)$$

$$= q\,C\,S\left[(C_L \cos \alpha + C_D \sin \alpha)\ (x - n) + (C_L \sin \alpha - C_D \cos \alpha)\ (y) - .8\ C_{L_t}\left(\frac{S_t}{S}\right) m\right] \quad (3a)$$

$$\text{or } \frac{M_{cg}}{q\,C\,S} = C_{M_{cg}} = \left[(C_L \cos \alpha + C_D \sin \alpha)\ (x - n)\right.$$
$$\left.(C_L \sin \alpha - C_D \cos \alpha)\ (y) - .8\ C_{L_t}\left(\frac{S_t}{S}\right) m\right] \quad (3b)$$

To determine whether or not the longitudinal stability is adequate, the degree of stability is determined by calculating the value $\left[\frac{d\ C_{M_{cg}}}{d\alpha}\right] / \frac{W}{S}$ the so-called Diehl's stability coefficient.

If therefore, the value of $C_{M_{cg}}$ is found for a series of angles of attack α, and plotted as the ordinate with values of α as the abcissa, then the value of the slope of the resulting curve, particularly in the

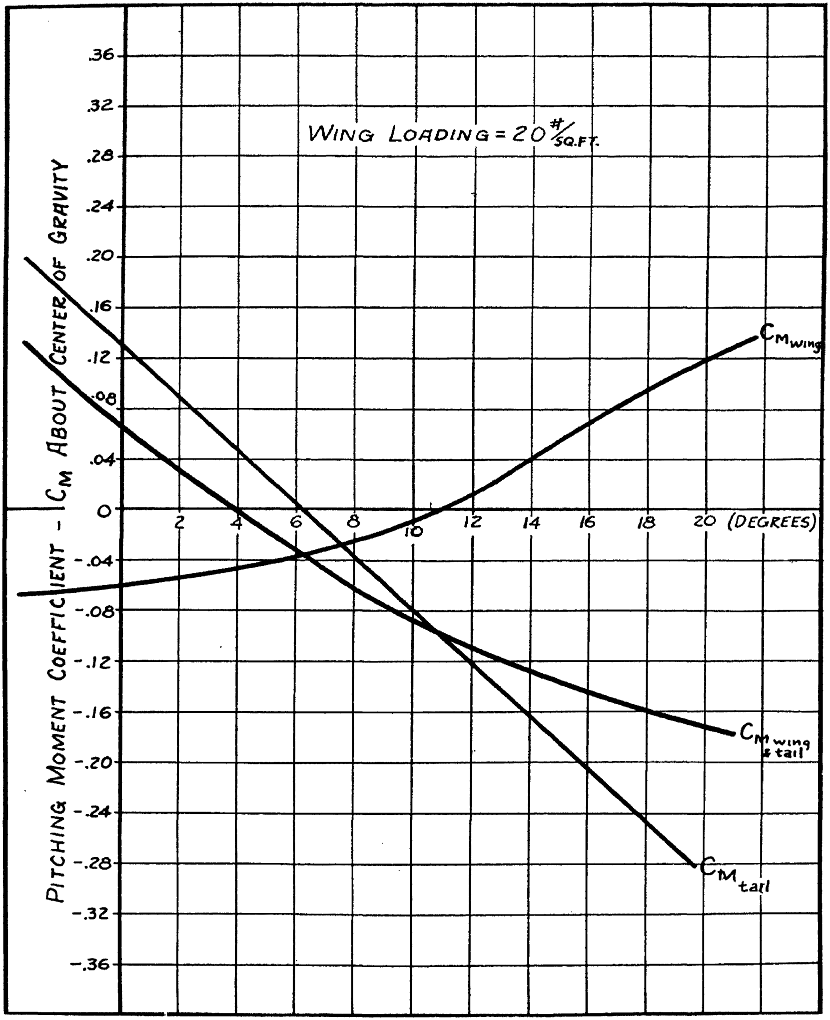


FIGURE 98. PITCHING MOMENTS ABOUT THE CENTER OF GRAVITY OF THE AIRPLANE FOR THE WING, TAIL, AND BOTH WING AND TAIL SURFACES

region in the angle of trim will give the value of $\frac{d\, C_{M_{cg}}}{d\, \alpha}$. This value divided by the wing loading W/S for the airplane will give the value of the stability coefficient.

The value, for commercial airplanes of the passenger transport type, of the stability coefficient lies between —.0005 and —.0010.

Angle of Trim and Horizontal Tail Surface Setting

At the angle of trim, the pitching moments about the center of gravity are zero by definition so that

$$M_{cg} = M_w + M_t = o$$

$$\text{or} \qquad M_t = -M_w \text{ or } \frac{M_t}{q\,C\,S} = \frac{-M_w}{q\,C\,S} \qquad (4)$$

Since an angle of trim for the airplane is desired either at the angle of attack at which the maximum speed or the cruising speed occurs, the moment of the forces acting on the wing about the center of gravity at the angle must equal the moments of the tail surfaces at that angle, but are opposite in sign.

The pitching moments of the aerodynamic forces on the wing may be calculated by means of equation (1b) above. This value gives the value of M_t, the required tail moment. Substituting in equation (2) above, the value of the required lift coefficient of the horizontal tail surfaces may be obtained. Then from calculations for downwash of the wing, and the airfoil characteristics of the horizontal tail surfaces, the angle of incidence of the horizontal tail surfaces may be obtained.

If the chord of the horizontal tail surfaces were at zero degrees to chord of the wing, then the tail surface would have the same angle of attack as the wing except for the downwash of the wing.

The general case, where both the wing and the tail surfaces have angle of incidence relative to the some fixed reference line such as the thrust line for instance, is shown in Figure 99.

i_w — represents the angle of incidence of the wing referred to the thrust line.

i_t — represents the angle of incidence of the horizontal tail surfaces referred to the thrust line.

α — angle of attack of the airplane referred to the line of the relative wind.

α_w = angle of attack of the wing referred to the line of the relative wind.

α_t = angle of attack of the horizontal tail surfaces referred to the new line of relative wind.

ϵ = angle of downwash.

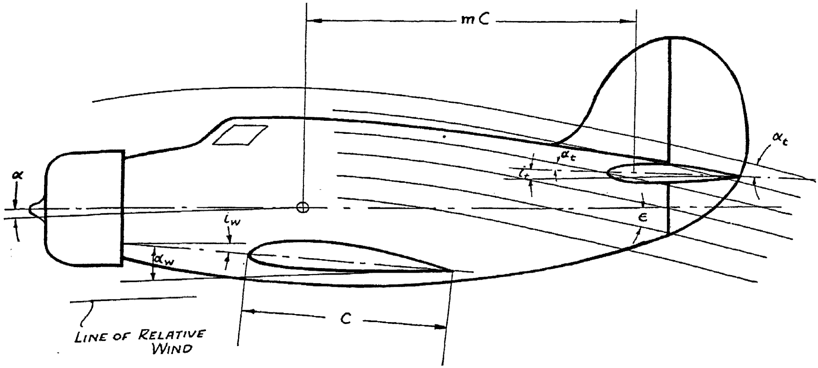


FIGURE 99

From Figure 99 it may be seen that:

$$\alpha_w = \alpha + i_w$$

$$\alpha_t = \alpha + \epsilon - i_t = \alpha_w - i_w + \epsilon - i_t$$

from which

$$i_t = \alpha_t - \alpha_w - \epsilon + i_w \qquad (5)$$

i_t and i_w are fixed angles

α_t, α_w, and ϵ vary with each attitude of the airplane.

The angle of attack of the wing is known since it corresponds to the required angle of trim; the angle of attack of the horizontal tail surfaces corresponds to the lift coefficient, C_{Lt}, required to obtain trim. The angle of downwash may be calculated from the empirical formula

$$= \left[\frac{3.8 - .32\ m}{R + 2}\right]\alpha \qquad (6)$$

where m (indicated in Figure 99) is the ratio of the tail length to the mean geometric chord of the wing.

R is the aspect ratio of the wing

and α_w is the angle of attack of the wing.

The lift coefficients (as well as the drag coefficients for preliminary performance calculation purposes) of the horizontal tail surfaces should be corrected for aspect ratio. The corrections depend upon planform and position as well as aspect ratio so that a few values for representative tail surfaces are given in Figure 100. These values were obtained by Benjamin F. Ruffner, Jr., and appeared originally in New York University Technical Note No. 1.

The lift coefficient for any angle of attack of the tail surfaces may then be determined from

$$C_{L_t} = \alpha_t \frac{d\,C_L}{d\,\alpha} \qquad (7)$$

For symmetrical airfoils, the angle of zero lift may be assumed to be at zero degrees angle of attack.

It is customary to set up a table for evaluating values of $C_{M_{cg}}$ in order to avoid chances of error in calculations and to reduce the amount of labor.

Formulae Required

(1) $C_{M_{cg}} = \left[(C_L \cos\alpha + C_D \sin\alpha)(x - n) + (C_L \sin\alpha - C_D \cos\alpha)(y) - .8\, C_{L_t}\left(\frac{S_t}{S}\right) m\right]$

(2) $\epsilon = \left[\frac{3.8 - .32\, m}{R + 2}\right] \alpha_w$

(3) $C_{L_t} = \alpha_t \frac{d\,C_L}{d\,\alpha}$

(4) $\alpha_t = \alpha_w - i_w + \epsilon - i_t$

DATA

Dimensions:— $x =$

$y =$

Angles:— $i_w =$

$i_t =$

Ratios:— $\frac{S_t}{S} =$

$\frac{W}{S} =$

$m =$

TABLE 31

1. α w	-4°	-2°	0°	2°	4°	6°	8°	10°	12°	14°	16°	18°	20°
2. C_L	(Corrected for aspect ratio)												
3. cos α													
4. C_L cos α													
5. C_D													

6. sin α
7. C_D sin α
8. (4) + (7)
9. n
10. (x — n)
11. (8) x (10)
12. C_L sin
13. C_D cos
14. (12) — (13)
15. (14) x (y)
16. (11) + (15)
17. ϵ [Determined for α_w listed in 1 and formula (2)]
18. α_t [Solution of formula (4)]
19. C_{L_t}
20. $C_{L_t}\, m \dfrac{S_t}{S}$
21. (16) — (20)

Items 1-17 pertain to the main wing.
Items 17-20 pertain to the horizontal tail surfaces.
Item 21 pertains to the complete airplane.

When this table has been completed the values of $C_{M_{cg}}$ given in line 21 are plotted against corresponding values of α_w given in line 1. The resulting curve will resemble that shown in Figure 101

The value of the $\dfrac{d\, C_{M_{cg}}}{d\ \alpha}$ obtained from the above graph, divided by the value of the wing loading, W/S, will give the value of the stability coefficient which, when compared with the criterion. determines whether the projected design will have sufficient static longitudinal stability.

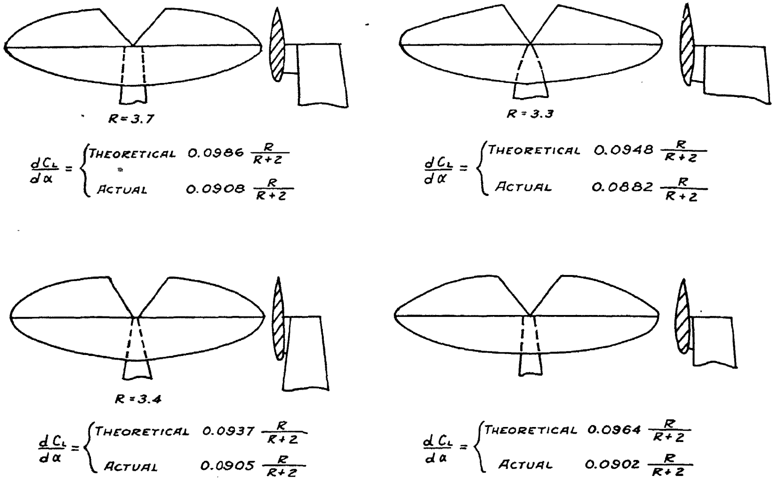


FIGURE 100 A. SLOPE OF LIFT CURVE FOR HORIZONTAL TAIL SURFACES PLACED ABOVE THE FUSELAGE

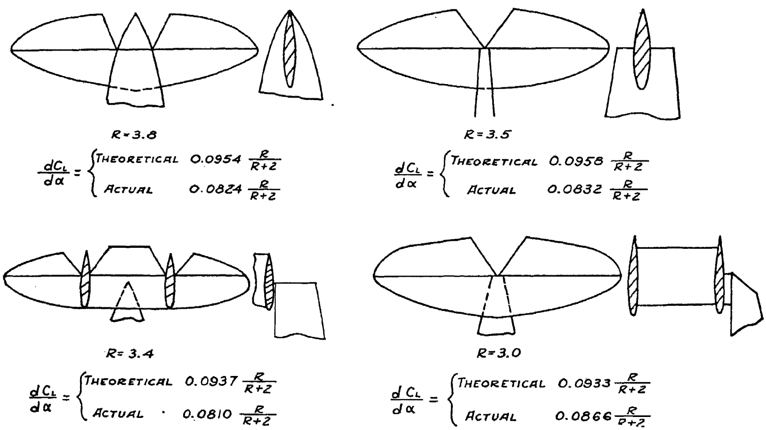
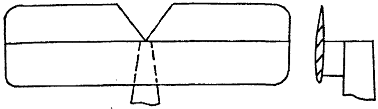
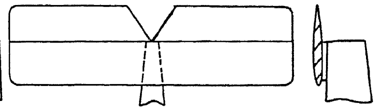


FIGURE 100 B. SLOPE OF LIFT CURVES FOR MISCELLANEOUS TYPES OF HORIZONTAL TAIL SURFACES



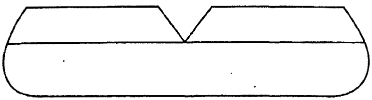
$R = 3.4$

$$\frac{dC_L}{d\alpha} = \begin{cases} \text{THEORETICAL} & 0.0937 \frac{R}{R+2} \\ \text{ACTUAL} & 0.0914 \frac{R}{R+2} \end{cases}$$

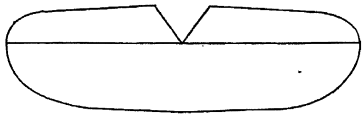


$R = 3.1$

$$\frac{dC_L}{d\alpha} = \begin{cases} \text{THEORETICAL} & 0.0955 \frac{R}{R+2} \\ \text{ACTUAL} & 0.0930 \frac{R}{R+2} \end{cases}$$

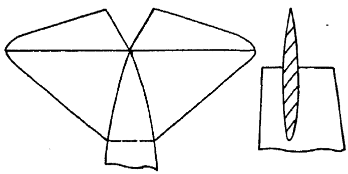


$$\frac{dC_L}{d\alpha} = 0.104 \frac{R}{R+2}$$



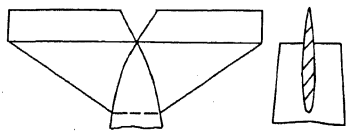
$$\frac{dC_L}{d\alpha} = 0.109 \frac{R}{R+2}$$

FIGURE 100 C. SLOPE OF LIFT CURVES FOR TYPICAL PLANFORMS OF HORIZONTAL TAIL SURFACES



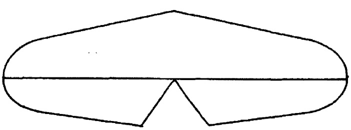
$R = 2.6$

$$\frac{dC_L}{d\alpha} = \begin{cases} \text{THEORETICAL} & 0.1043 \frac{R}{R+2} \\ \text{ACTUAL} & 0.0743 \frac{R}{R+2} \end{cases}$$

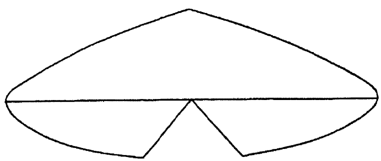


$R = 3.5$

$$\frac{dC_L}{d\alpha} = \begin{cases} \text{THEORETICAL} & 0.0958 \frac{R}{R+2} \\ \text{ACTUAL} & 0.0581 \frac{R}{R+2} \end{cases}$$

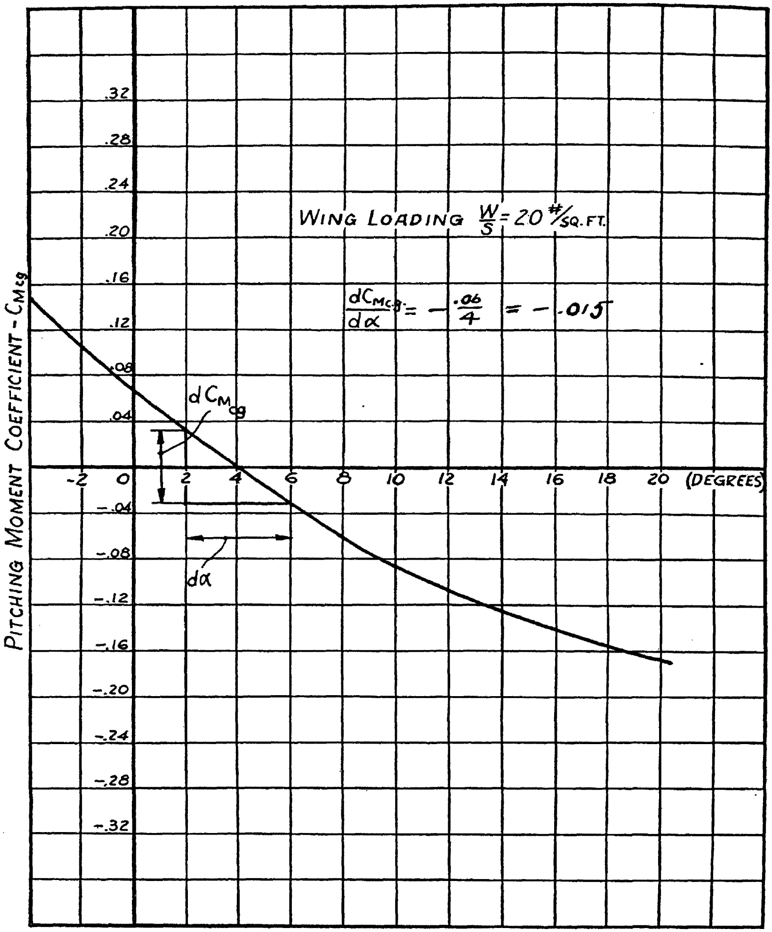


$$\frac{dC_L}{d\alpha} = 0.108 \frac{R}{R+2}$$



$$\frac{dC_L}{d\alpha} = 0.107 \frac{R}{R+2}$$

FIGURE 100 D. SLOPE OF LIFT CURVES FOR GENERALLY POOR PLANFORMS OF HORIZONTAL TAIL SURFACES



WING LOADING $\frac{W}{S}$ = 20 #/SQ.FT.
$\frac{dC_{Mc.g.}}{d\alpha} = -\frac{.06}{4} = -.015$
$dC_{M_{cg}}$
$d\alpha$
PITCHING MOMENT COEFFICIENT - C_{Mcg}
(DEGREES)

FIGURE 101

XVI. Preliminary Performance Calculations

Performance calculations are necessary at several stages of the design to determine whether or not the original specifications set forth for the design are being met. The first set of preliminary performance calculations is based upon empirical formulae that have been established on the basis of past performance of similar airplanes and are predicated upon similar power and wing loadings, gross weights and general design. These calculations are wholly rule-of-thumb and are used by the engineer for his general information and to decide whether his design is likely to meet the performance specifications at all.

The second form of preliminary performance calculations is set forth briefly in this chapter. The data for determining the horsepower required to fly in horizontal flight may be obtained in several ways:

1. From scattered wind tunnel tests made by various testing agencies.
2. From wind tunnel tests on individual components of the design.
3. From wind tunnel tests on the complete design.
4. From wind tunnel tests and full flight corrections obtained on similar previous designs.

These calculations can be considered only as preliminary since their accuracy depends upon the judgment of the engineer using the particular data.

Performance data must be obtained finally from actual full flight tests.

Preliminary performance estimates for minimum or stalling speed, maximum speed, maximum rate of climb, service and absolute ceiling and range may be obtained by a series of calculations for horsepower required at various speeds and the corresponding horsepower available. These calculations are given here in briefest form, and their accuracy depends upon the accuracy of the various assumptions made. However, they may be considered sufficiently accurate until extensive wind tunnel tests on the design have been made.

Horsepower Required

The horsepower required at different speeds in horizontal flight is calculated by means of the equation:

$$P_r = (D_w + D_p)\frac{V}{550} \quad \text{where}$$

D_w is the wing drag in pounds at speed V,
D_p is the sum of parasite and interference drag in pounds at speed V,
V is the speed in feet per second of the airplane in horizontal flight at the corresponding angle of attack.

The wing drag coefficients are usually obtainable for an aspect ratio 6 and must be corrected for the aspect ratio used. These corrections are outlined in Chapter XI, Table 27.

The parasite resistance includes all the resistance caused by the fuselage, tail surfaces, surface irregularities, landing gear, supports and the like—in other words all those parts of the airplane except the wing proper.

Parasite resistance coefficients are obtainable from various sources and are often referred to unit length as in the case of struts, or per unit as in the case of wheels, or per square foot of maximum cross section in the case of the fuselage. It is usually desirable to obtain the total drag for a speed of one foot per second and then obtain a new coefficient referred to the wing area by dividing by the wing area.

The interference drag is more difficult to appraise. Usually it is accounted for in the calculations by arbitrarily increasing the parasite resistance by a small percentage. An airplane with a fuselage well faired into the wing, with landing gear retracted, offers far less interference drag than one not so well faired. The arbitrary factor may be as much as 5 per cent.

These calculations are best carried out by means of a table, such as is outlined below:

1	2	3	4	5	6	7	8	9	10	11	12	13	14
α	D_f	D_{Lg}	$D_{V.T.S.}$	D_{HTS}	D_p	$D_p \times 1.05$	C_{D_p}	C_{D_w}	C_{D_t}	C_L	v	D	HP

Column 1—Here are entered the angles of attack of the airplane or the wing, whichever is chosen for reference. The angular range should start slightly above the angle at which zero lift of the wing occurs and stop at a few angles beyond that at which maximum lift of the wing occurs. It is usually satisfactory to use 2-degree intervals.

Column 2—Record here the total drag, at 1 foot per second, of the fuselage. The fuselage drag usually varies with angle of attack. If this change is considered small, it may be neglected and thus be assumed constant for all angles of attack.

Column 3—The total drag of the landing gear, at 1 foot per second, of the wheels, struts and cowling. This drag usually remains constant for all angles of attack.

Column 4—The vertical tail surfaces do not change their angle of attack as the airplane changes its angle of attack; therefore the drag remains constant for all angles of attack. Any blanketing effect of the fuselage on the vertical tail surfaces is neglected. The drag of the vertical tail surfaces at one foot per second is therefore constant.

Column 5—The drag of the horizontal tail surfaces changes with angle of attack so that the drag at 1 foot per second varies with increasing angle of attack of the airplane. The drag coefficients of the horizontal tail surfaces must be corrected for aspect ratio. These corrections are outlined in Chapter XV. Usually no account is taken of difference in the drag of the horizontal tail surfaces due to downwash of wing.

Column 6—If columns 2, 3, 4 and 5 account for all items (outside of the wing) contributing to the parasite drag, then the addition of these columns for each angle of attack will give the total parasite resistance, designated D_p, at 1 foot per second.

Column 7—As previously explained, the interference drag may be accounted for by multiplying the parasite resistance by a factor varying from 1.00 to 1.05. This may be called a "bogey factor" by which any omissions in the calculation of parasite resistance may readily be accounted for if so desired.

Column 8—The value of C_{D_p}, the parasite and inference resistance coefficient in the absolute system referred to the wing area, may be found by dividing the values found in column 6 by $\frac{1}{2} \rho S_w v^2$ where S_w represents the wing area and since $v = 1$ f.p.s. (the speed for which calculations were made, or if desired, calculations may be made for 100 f.p.s.) the expression becomes $\frac{1}{2} \rho S_w$.

Column 9—Here are recorded the drag coefficients of the wing corrected for aspect ratio, of the airfoil used in the design corresponding to the angles of attack listed in column 1.

Column 10—The addition of the C_{D_p} and C_{D_w} gives C_{D_t} the total resistance coefficient of the airplane referred to the wing area.

Column 11—The lift coefficient for the airfoil used for the wing, for the angles of attack given in column 1, are listed here.

Column 12—The speed of the airplane for increasing angle of attack in horizontal flight is found by means of the formula

$$v \text{ (f.p.s.)} = \sqrt{\frac{\text{Gross Weight}}{\frac{1}{2}\rho C_L S}}$$

which for a given design may be simplified to the form of

$$v \text{ (f.p.s.)} = K\sqrt{\frac{1}{C_L}}$$

If desired, the expressions $1/C_L$ and $\sqrt{1/C_L}$ may be included in the above table or included in a subsidiary table. The values for C_L for different angles of attack are those listed in column 11.

Column 13—The total drag of the airplane is then found by means of the expression $D = \frac{1}{2}\rho S C_{D_t} v^2$ which for a given design may be simplified to read $D = K_1 C_{D_t} v^2$ where corresponding values of C_{D_t} and v may be found in columns 10 and 12 respectively.

Column 14—The horsepower required, H.P., is found by means of the formula

$$HP = \frac{D\ v}{550}$$

where D and corresponding v are found in columns 13 and 12 respectively.

The above comprises the calculations for horsepower required. It is customary to plot the values of HP versus corresponding speed (usually given in miles per hour), then with the aid of horsepower available curves, almost all the performance figures required may be calculated readily.

Altitude Calculations

The above calculations as outlined are usually calculated for sea level conditions. They may be extended to include calculations

for other altitudes by multiplying by the square roots of the relative densities both the velocity and horsepower found at another altitude as follows:

$$v_1 = v_0 \sqrt{\frac{\rho_0}{\rho_1}}$$

and

$$HP_1 = HP_0 \sqrt{\frac{\rho_0}{\rho_1}}$$

where v_1 represents the new speed at the density ρ_1 at the new altitude,
v_0 represents the originally calculated speed at the density ρ_0,
HP_1 represents the new horsepower corresponding to v_1,
HP_0 represents the originally calculated horsepower corresponding to v_0.

The density relationships of any altitude to that of sea level may be found in Table 32.

TABLE 32

Altitude in Feet	t — °F	ρ	$\frac{\rho}{\rho_0}$	$g\rho$ *Lb./ft.*³	p *Inches*	$\frac{p}{p_0}$
0	59.	.002378	1.	.07651	29.92	1.
1000	55.434	.002309	.9710	.07430	28.86	.9644
2000	51.868	.002242	.9428	.07213	27.82	.9298
3000	48.301	.002176	.9151	.07001	26.81	.8962
4000	44.735	.002112	.8881	.06794	25.84	.8636
5000	41.169	.002049	.8616	.06592	24.89	.8320
6000	37.603	.001988	.8358	.06395	23.98	.8013
7000	34.037	.001928	.8106	.06202	23.09	.7716
8000	30.471	.001869	.7859	.06013	22.22	.7427
9000	26.904	.001812	.7619	.05829	21.39	.7147
10000	23.338	.001756	.7384	.05649	20.58	.6876
11000	19.772	.001702	.7154	.05474	19.79	.6614
12000	16.206	.001648	.6931	.05303	19.03	.6359
13000	12.640	.001596	.6712	.05136	18.28	.6112
14000	9.074	.001545	.6499	.04973	17.57	.5873
15000	5.507	.001496	.6291	.04814	16.88	.5642
16000	1.941	.001448	.6088	.04658	16.21	.5418
17000	—1.625	.001401	.5891	.04507	15.57	.5202
18000	—5.191	.001355	.5698	.04359	14.94	.4992
19000	—8.757	.001311	.5509	.04216	14.33	.4790
20000	—12.323	.001267	.5327	.04075	13.74	.4594
21000	—15.890	.001225	.5148	.03938	13.18	.4405
22000	—19.456	.001183	.4974	.03806	12.63	.4222
23000	—23.022	.001143	.4805	.03676	12.10	.4045

TABLE 32—*Continued*

Altitude in Feet	t — °F	ρ	$\frac{\rho}{\rho_0}$	$g\rho$ Lb./ft.³	p Inches	$\frac{p}{p_0}$
24000	—26.588	.001103	.4640	.03550	11.59	.3874
25000	—30.154	.001065	.4480	.03427	11.10	.3709
26000	—33.720	.001028	.4323	.03308	10.62	.3550
27000	—37.287	.000992	.4171	.03192	10.16	.3397
28000	—40.853	.000957	.4023	.03078	9.72	.3248
29000	—44.419	.000922	.3869	.02968	9.293	.3106
30000	—47.985	.000889	.3740	.02861	8.880	.2968
31000	—51.551	.000857	.3603	.02757	8.483	.2834
32000	—55.117	.000826	.3472	.02656	8.101	.2707
33000	—58.684	.000795	.3343	.02558	7.732	.2583
34000	—62.250	.000765	.3218	.02463	7.377	.2465
35000	—65.816	.000736	.3098	.02367	7.036	.2352
36000	—67.	.000704	.2962	.02265	6.708	.2242
37000	—76.	.000671	.2824	.02160	6.394	.2137
38000	—67.	.000640	.2692	.02059	6.096	.2037
39000	—67.	.000610	.2566	.01963	5.812	.1943
40000	—67.	.000582	.2447	.01872	5.541	.1852
41000	—67.	.000554	.2332	.01785	5.283	.1765
42000	—67.	.000529	.2224	.01701	5.036	.1683
43000	—67.	.000504	.2120	.01622	4.802	.1605
44000	—67.	.000481	.2021	.01546	4.578	.1530
45000	—67.	.000459	.1926	.01474	4.365	.1458
46000	—67.	.000437	.1837	.01405	4.160	.1391
47000	—67.	.000417	.1751	.01339	3.966	.1325
48000	—67.	.000397	.1669	.01277	3.781	.1264
49000	—67.	.000379	.1591	.01217	3.604	.1205
50000	—67.	.000361	.1517	.01161	3.436	.1149

Parasite Resistance Data

The data on parasite resistance may be found from wind tunnel tests and N.A.C.A. reports, as well as in various handbooks. The data given below are representative.

Item	*Full scale Drag in Lb. at 100 m.p.h.*
Vertical tail surface cantilever per square foot. If the minimum drag coefficient of the vertical tail surfaces is known, this value may be used with an increase of about 25 per cent to account for gap at hinges. If the surfaces are externally braced, increase the drag by about 50 per cent.	0.40
Horizontal tail surfaces, cantilever, use drag coefficients, corrected for aspect ratio, for each angle of attack. Add about 25 per cent for gap at hinges. If horizontal tail surfaces are externally braced, increase the drag coefficients by about 50 per cent. If the external bracings are struts of streamlined tubing, allow additional foot for each end fitting, and estimate drag of bracing on basis of projected length of these struts when thrust line is horizontal.	
Fuselage plus cowled engine per square foot of projected maximum frontal area. If the fineness ratio (ratio of length to depth) is rather small, use high figure. The variation of fuselage drag (for approximately elliptical cross section) may be obtained by means of the formula	6-8.5

$$D_\alpha = D_o + 0.0015\, D_o\, \alpha$$

α = angle of attack in degrees

where D_o = value given above (for $\alpha = 0$ degrees)

The Engine

The horsepower available at different speeds of the airplane in horizontal flight is determined from the brake horsepower curves of the engine used for the design, and the propeller characteristics.

The brake horsepower, usually plotted against corresponding revolutions per minute of the engine is generally furnished by the manufacturer. This power curve is given for sea level conditions unless the engine happens to be supercharged when the power curve corresponds to the rated altitude limit of the supercharger.

The brake horsepower of non-supercharged engines and of supercharged engines above the rated altitude limit of the supercharger varies with the altitude. This variation may be determined easily from the following formula:

$P = P_o \left(\frac{\rho}{\rho_o}\right)^{1.2}$ where

P = the brake horsepower at altitude considered.

P_o = the brake horsepower at sea level or at limiting supercharged altitude

ρ = density at altitude under consideration.
ρ_0 = the density at sea level or at limiting supercharged altitude.

The following should also be borne in mind when the engine horsepower is desired for other than the usual standard for the engine in question.

1. Propeller gearing will reduce the engine brake horsepower 2 or 3 per cent.
2. For engines equipped with turbo supercharger, the power decreases to about 80 per cent of the normal sea level power of the unsupercharged engine.
3. For direct drive superchargers, the power at the limiting altitude of the supercharger is the same as the normal sea level power developed by the unsupercharged engine.

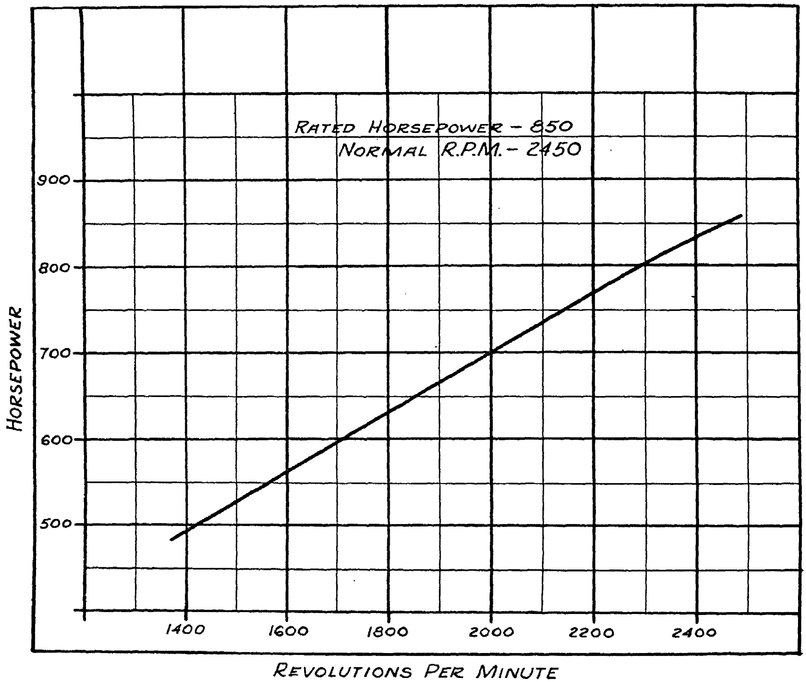


FIGURE 102. VARIATION OF BRAKE HORSEPOWER OF ENGINE WITH REVOLUTIONS PER MINUTE

A typical power curve for an engine is shown in Figure 102.

Propeller Selection

In selecting a propeller, the maximum speed as well as the rated horsepower at normal r.p.m. are the design conditions.

Since at maximum speed, the thrust horsepower of the engine-propeller unit equals the horsepower required in horizontal flight, and since

$$T.H.P._m = B.H.P._m \times \eta_m$$

where $T.H.P._m$ is the maximum thrust horsepower

$B.H.P._m$ is the normal brake horsepower of the engine at rated r.p.m.

η_m is maximum efficiency of the propeller,

it is easy, by means of a few trials to determine the required propeller diameter.

Example: The horsepower required for corresponding speeds has

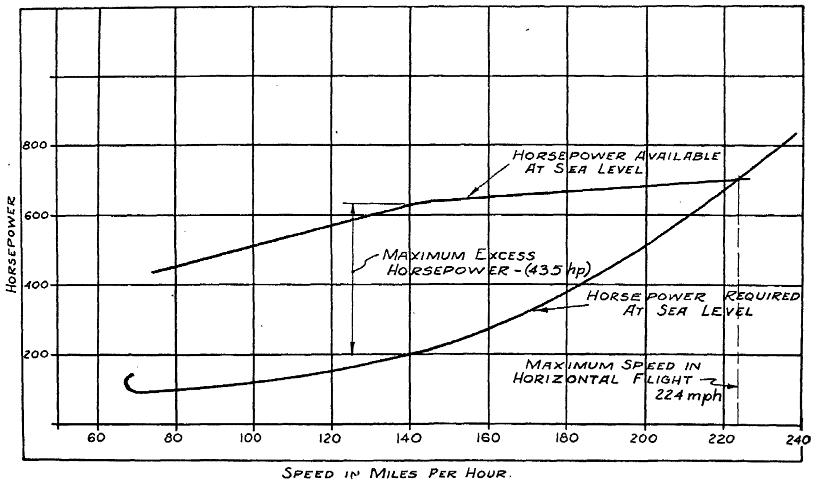


FIGURE 103

been calculated for a given design and plotted as shown in Figure 103. The engine chosen will deliver 850 horsepower at 2450 r.p.m. (crankshaft speed) at sea level conditions. The propeller is geared at 3 to 2 ratio, so that the propeller speed is 2/3 of the crankshaft

speed or will rotate at 1634 r.p.m. The brake horsepower is given for the geared engine so that it is unnecessary to allow for the 2 or 3 per cent reduction in power due to gearing.

First assume propeller efficiency at maximum speed = 80 per cent. This gives a thrust horsepower of 850 x .8 = 680 horsepower. Referring to Figure 103, it is found that for a required horsepower of 680, the speed in horizontal flight would be 222 m.p.h. at sea level.

Next the value of C_S for the propeller is calculated. It is a convenient "power coefficient" which is used in an equation not involving the propeller diameter.

$$C_S = \frac{2.14 \ V_{mph}}{(BHP)^{1/5} \ (N_{rpm})^{2/5}} \quad \sqrt[5]{\rho}$$

For the conditions set forth.

$$C_S = \frac{2.14 \ (V_{mph})}{(850)^{1/5} \ (1634)^{2/5}} \quad \sqrt[5]{.002378} = .008611 \, V_{mph}$$

For 222 m.p.h. $C_S = 1.91$.

The corresponding maximum efficiency as obtained from Figure 104 is 86 per cent, indicating that our initial assumption of 80 per cent for the propeller efficiency was somewhat low. After a few more trials the efficiency of 86 per cent seems to give the closest results and the corresponding high speed is 227 m.p.h.

When exact agreement has been obtained in the assumed and extrapolated values for the propeller efficiency, it is then possible to calculate the correct propeller diameter.

Referring to Figure 104, point A represents $C_S = 1.95$, project a line upwards until it intersects the dotted line marked "line of maximum efficiency for given C_S" at point B, then project a line at right angles toward the right until it intersects the scale for V/nD at point C. The value of V/nD at point C is 1.18. Since V and n are known, D, the diameter of the propeller, may be solved for as follows:

$$\frac{V}{nD} = 1.18 = \frac{(227 \times 1.467)}{\left(\frac{1634}{60}\right) D}$$

D = 10.35 feet. Say 10 feet, 6 inches. The blade angle at 75 per cent of the radius is about 30 degrees.

The critical propeller tip speed is about 1000 feet per second and it should not be exceeded if serious reduction in propeller efficiency is to be avoided. The critical propeller tip speed, V_c, may be calculated from the following formula:

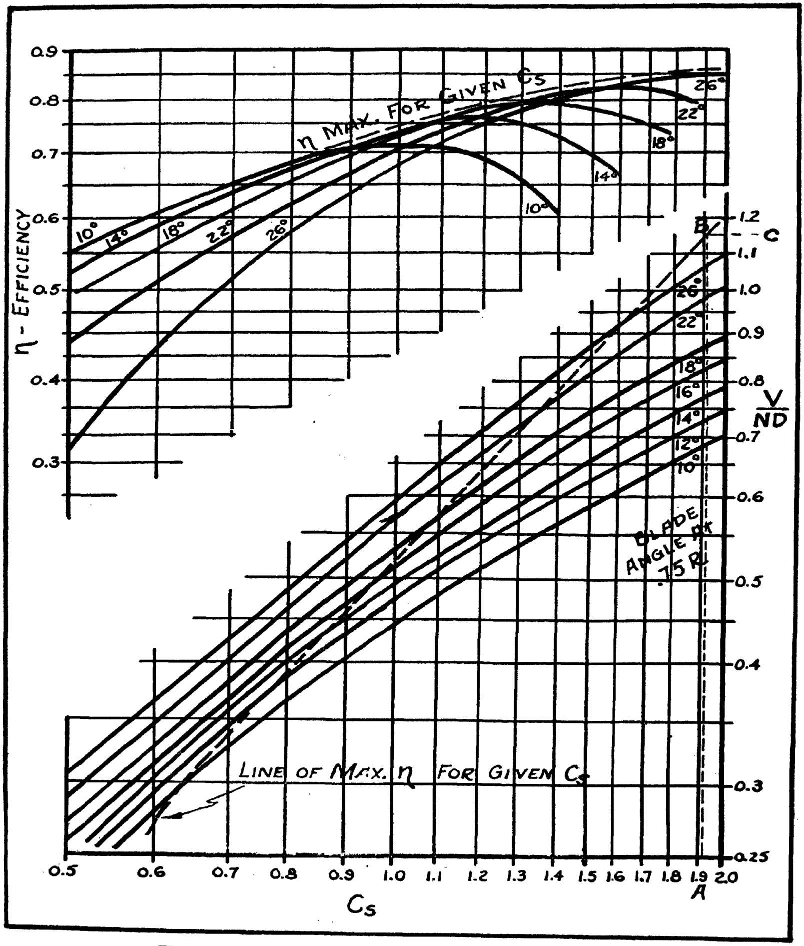


FIGURE 104. PROPELLER CHARACTERISTICS

$$V_c = \sqrt{\left(\frac{\pi\, DN_{r.p.m.}}{60}\right)^2 + (V)^2}$$

For the propeller diameter just calculated V_c becomes equal to

$$\sqrt{\left[\frac{\pi\,(10.5)\,(1634)}{60}\right]^2 + (227 \times 1.467)^2} = 960 \text{ f.p.s.}$$

The propeller should therefore give excellent performance at sea level conditions. If best cruising speed is desired at another altitude, say that to which the engine is supercharged, then the corresponding values of brake horsepower and air density must be substituted in the equation given for calculating the value of the power coefficient C_s.

Figure 104 represents propeller characteristics when the propeller is placed in front of a cowled radial engine located in the leading edge of a moderately thick wing. These characteristics depend upon obstructions in the propeller slipstream. If more accurate values for efficiency and blade angles are desired, pertinent N.A.C.A. reports should be consulted.

If a two-bladed propeller has too large a diameter so that the "tip" speed is too high or there is insufficient ground clearance or other factors prevailing which make it necessary to choose a smaller diameter, a 3-bladed propeller may be required.

For the conditions set forth, again assuming an initial efficiency of 80 per cent:

$$C_s = \frac{2.14\ (V_{m.p.h.})}{(850)^{1/5}\ (1634)^{2/5}} \qquad \sqrt[5]{.002378} = .0086\ V$$

For 222 m.p.h. $C_s = 1.91$.

The corresponding maximum efficiency as found before was 86 per cent. This efficiency is for a two-bladed propeller whereas a three-bladed propeller is to be used, the maximum efficiency of which would be reduced from 86 to 83 per cent, which is still higher than initially assumed.

After successive trials, it is found that maximum speed is closer to 224 miles an hour, for which the value of $C_s = 1.92$, and the propeller efficiency for a two-bladed propeller is about 86 per cent, or 83 per cent for a three-bladed propeller.

For this value of C_s, the corresponding V/nD, determined as before, is 1.15

$$\frac{V}{nD} = 1.15 = \frac{224 \times 1.467}{\frac{(1634)}{60} D}$$

D = 10.47, say 10 feet 6 inches. The blade angle at 75 per cent of the radius is approximately 29.5 degrees. Again this value is for a two-bladed propeller since all values have been based upon those obtained from Figure 104 which is for a two-bladed propeller. However these same values may be used, and the diameter so found may be reduced 7 per cent to obtain the corresponding diameter of a three-bladed propeller. The diameter of the three-bladed propeller is, then, 10.5 feet less 7 per cent or 9 feet 9 inches, roughly.

Diameters and ratings of typical controllable pitch propellers are listed in the table below:

TABLE 33

Hamilton Standard Propeller Data

No. of Blades	*Diameter*	*R.P.M. (Max.)*	*Angular Range in Degrees*	*Weight in Pounds*	*Type of Engine with Which It May Be Used*
2	9′10″	2200	8°	103.5	Continental, Jacobs, Lycoming, Whirlwind, 7 cylinder direct drive.
2	10′0″	2200	10°	160.0	Twin Wasp Jr. direct drive Wasp & Wasp Jr. direct drive Whirlwind 9 cylinder.
2	9′6″	2400	10°	154.0	Same.
3	10′0″	2200	10°	240.5	Wasp and Twin Wasp geared Whirlwind 2 row geared.
3	9′6″	2400	10°	231.5	Same.
2	10′0″	2200	10°	172.0	Wasp H Direct Drive.
2	9′6″	2400	10°	166.	Same.
2	11′6″	1680	10°	217.5	Hornet Direct Drive Twin Wasp Jr. geared Whirlwind 2 row geared Cyclone Direct Drive.
2	10′6″	2100	10°	214.5	Same.
3	11′6″	1680	10°	337.0	Twin Wasp geared 3:2 Hornet geared Cyclone geared 16:11.
3	13′0″	1500	10°	355.	Twin Wasp geared 3:2 Hornet geared.

PRELIMINARY PERFORMANCE CALCULATION

Lycoming Smith Controllable Propeller

9′6″	1950	289.4	735 HP
11′0″	1340	404.8	750 HP

Alternative Method of Determining Propeller Diameter

The diameter required for the two-bladed propeller may be calculated directly by means of the formula

$$\text{Propeller diameter} = \frac{67}{\sqrt[4]{\rho}} \times \sqrt[4]{\frac{BHP}{V_{m.p.h.}}} \times \sqrt{\frac{1}{N_{r.p.m.}}}$$

For the example cited

$$D = \frac{67}{\sqrt[4]{.002378}} \times \sqrt[4]{\frac{850}{224}} \times \sqrt{\frac{1}{1634}} = 10.45 \text{ feet.}$$

The same formula may be used to calculate the three-bladed propeller by using only 70 per cent of the value for the brake horsepower in the formula. The value of C_s for a three-bladed propeller may also be calculated on the basis of 70% the brake horsepower.

Horsepower Available

The modern constant speed propellers are so designed as to keep the same revolutions per minute regardless of the forward speed of the airplane—at least within the angular range of the propeller blades. It is easy therefore to determine the horsepower available by calculating the value of C_s and determining the corresponding efficiency and blade angle. It is desirable to determine the blade angle also since the constant speed propellers are usually limited to 8 to 10 degrees in angular range. The method of determining the horsepower available may be illustrated best by the results of calculations presented in the table below.

TABLE 34

Design Speed 224 m.p.h.
Propeller Diameter 9.75, 3 blades
Engine 850 horsepower at 2450 r.p.m.

Sea Level = .002378
Gear Ratio 2:3

TABLE 34—*Continued*

1	2	3	4	5	6	7	8	9
V (m.p.h)	$\frac{V^*}{nD}$	C_S	β^*	η^*	η_1	*Propeller R.P.M.*	*Brake HP*	*HP Available*
224	1.11	1.92	29.5°	86	83	1634	850	706
200	1.025	1.72	28.5	85	82	1634	850	697
180	.922	1.55	27.0	83	80	1634	850	680
160	.821	1.375	26.5	81	78	1634	850	664
140	.717	1.202	25.5	78	75	1634	850	638
120	.616	1.032	25.0	74	71	1634	850	604
100	.512	.860	24.5	69	66	1634	850	562
80	.410	.688	24.0	64	61	1634	850	518
60	.307	.517	24.0	58	55	1634	850	467

* Based upon two-bladed propeller.

Column 1—Speeds are tabulated in this column usually in 20-mile intervals between approximate stalling speed to maximum speed.

Column 2—Values of $\frac{V}{nD}$ are calculated for speeds listed in column 1.

Column 3—Values of C_S are calculated for the values of speeds tabulated in column 1, and horsepower and corresponding r.p.m. values tabulated in columns 8 and 7, respectively.

Column 4—The blade angles at 75 per cent of the radius are determined for the values of C_S given in column 2 with the aid of Figure 104.

Column 5—The propeller efficiency corresponding to the calculated values of C_S and blade angle is determined with the aid of the curves at the top of Figure 104 as previously explained.

Column 6—Since the efficiencies tabulated in column 5 are for two-bladed propellers, a subtraction of 3 per cent is made from those values to give the efficiencies for a three-bladed propeller. See column 6.

Columns 7 and 8—Since the controllable pitch propeller is a constant speed propeller, the engine brake horsepower and corresponding revolutions per minute remain constant throughout the angular range.

Column 9—The efficiency of a two-bladed propeller for the values of the corresponding propeller efficiencies given in column 6 give the horsepower available at full throttle at the speed listed in column 1. These values are plotted in Figure 106.

In case of a fixed pitch propeller, the calculations are slightly different since the engine r.p.m. and power do not remain the same. Suppose the controllable pitch propeller used above could not change

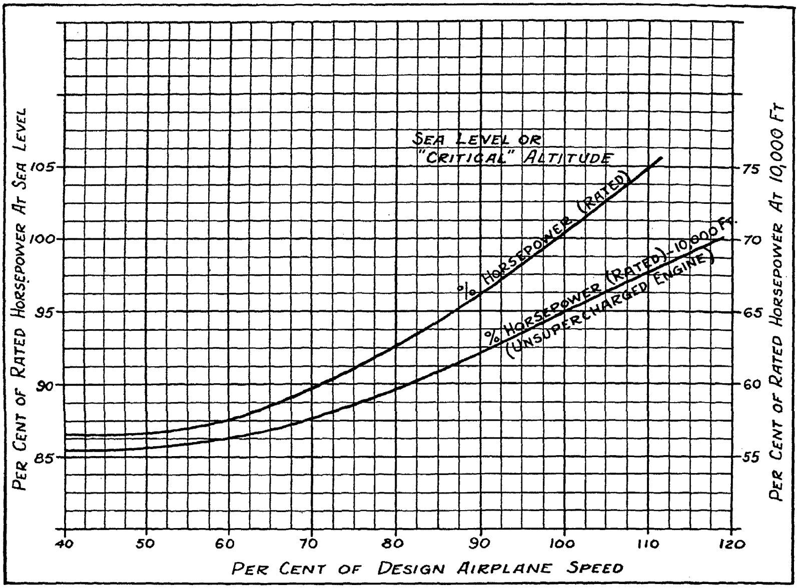


FIGURE 105. POWER VARIATION OF AIRCRAFT ENGINE WITH CHANGE IN SPEED WHEN A FIXED PITCH PROPELLER IS USED

TABLE 35

1	2	3	4	5	6	7	8	9	10	11
V	% of Design Speed	% of Rated HP	Brake HP	R.P.M.	Propeller R.P.M.	C_S	β	η*	η_1	HP Available
140	100%	100%	850	2450	1634	1.202	25.5	78	75	638
120	85.7	94.5	803	2300	1535	1.071	25.5	74	71	570
100	71.4	90.0	765	2185	1456	.919	25.5	72	69	528
80	57.2	87.2	742	2120	1415	.785	25.5	67	64	474
60	42.8	86.5	727	2080	1387	.570	25.5	58	55	400

* Based upon two-bladed propeller.

Column 1—The speeds for the fixed pitch range of the propeller are tabulated in this column.

Column 2—The per cent of design speed are calculated for the speeds given in column 1. These values are used in conjunction with Figure 105. Since the propeller is assumed to become a fixed pitch propeller at about 140 m.p.h., the design speed is then 140 m.p.h. Other speeds are then a certain per cent of this design speed.

Column 3—The per cent of rated power for corresponding per cent of design speed has been found with the aid of the curve in Figure 105, indicated as the per cent of rated horsepower at sea level or critical altitude.

Column 4—By multiplying the rated horsepower by the percentages tabulated in column 3, the actual brake horsepower corresponding to the speeds tabulated in column 1 has been found. The rated horsepower is that horsepower, in this case, of the engine as it enters the fixed pitch propeller range.

Column 5—The engine r.p.m. corresponding to the brake horsepower tabulated in column 4 may be found with the aid of Figure 102.

Column 6—Since the propeller gear ratio is 3:2, the actual propeller r.p.m. values have to be calculated.

Column 7—The values of C_S are calculated for the corresponding values of speed, power, and propeller r.p.m. found in columns 1, 4 and 6.

Column 8—The propeller blade angle remains constant.

Column 9—The efficiency of a two-bladed propeller fo the values of C_S in column 7 and $\beta = 25.5°$ are obtained from the top part of Figure 104.

Column 10—Reduction is made for the three-bladed propeller.

Column 11—Multiplying the values of brake horsepower in column 4 by the corresponding efficiencies, η_1, in column 10 will give the values of horsepower available in column 11.

The calculations for other altitudes are similar to those given for sea level conditions. The horsepower-available values for the critical altitude of the supercharged engine are tabulated in Table 36.

The upper limit of the controllable pitch propeller of 29.5 degrees has been determined by the design sea level condition and therefore remains the same.

TABLE 36

Design Speed 238 m.p.h. Propeller Diameter 9.75 feet, 3 blades
Altitude 7000 $\rho = .001928$ Gear Ratio 2:3
Engine 850 horsepower @ 2450 r.p.m.

V m.p.h.	C_S	$\frac{V}{nD}$*	β*	η*	η_1	Prop. R.P.M.	Brake HP	HP Available
238	1.982	1.22	29.5	86	83	1634	850	705
220	1.815	1.13	29.0	85	82	1634	850	697
200	1.65	1.025	28.5	84	81	1634	850	687
180	1.485	.922	28.0	83	80	1634	850	680
160	1.321	.821	27.5	82	79	1634	850	672
145	1.20	.744	27.0	79	76	1634	850	646

* Based upon two-bladed propeller.

The horsepower-available and the horsepower-required curves at the critical altitude of 7000 feet have been plotted in Figure 106.

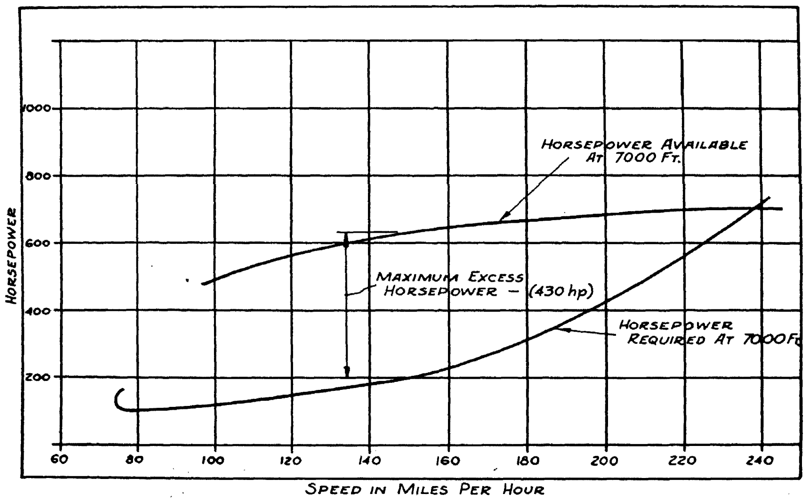


FIGURE 106

Above the supercharged altitude, the engine develops less than its rated horsepower. This variation with altitude above the critical altitudes for supercharged engines is shown in Figure 107.

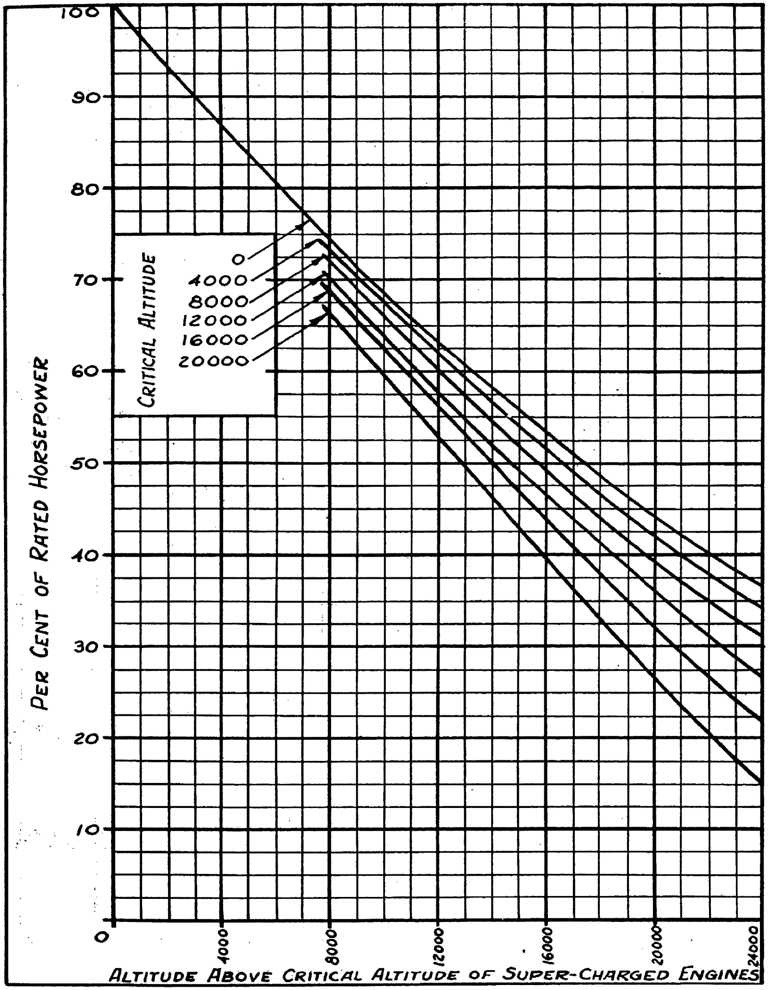


FIGURE 107. VARIATION OF BRAKE HORSEPOWER OF A SUPER CHARGED ENGINE ABOVE THE CRITICAL ALTITUDE

The rated horsepower at 10,000 feet for the engine in the problem just illustrated is only 89 per cent of that at its critical altitude of 7000 feet, or .89 x 850 = 757 hp. The corresponding r.p.m. (from Figure 102) is 2165 r.p.m. The calculations for horsepower available are otherwise similar to those in Tables 34 and 36.

The stalling speed may be determined from the formula

$$= \sqrt{\frac{\text{Gross Weight}}{\frac{1}{2}\rho \, C_{L_{max}} \, S}}$$

The landing speed is usually assumed to be equal to, or a few miles less than, the calculated stalling speed.

Maximum Speed

The intersection of the curves of the horsepower available and the horsepower required determines the maximum speed, in horizontal flight of the airplane. In Figure 103, the maximum speed at sea level is 224 miles per hour. Figure 106 indicates that the maximum speed at 7000 feet is 238 miles per hour, and Figure 108 indicates that the maximum speed at 10,000 feet is 237 miles per hour.

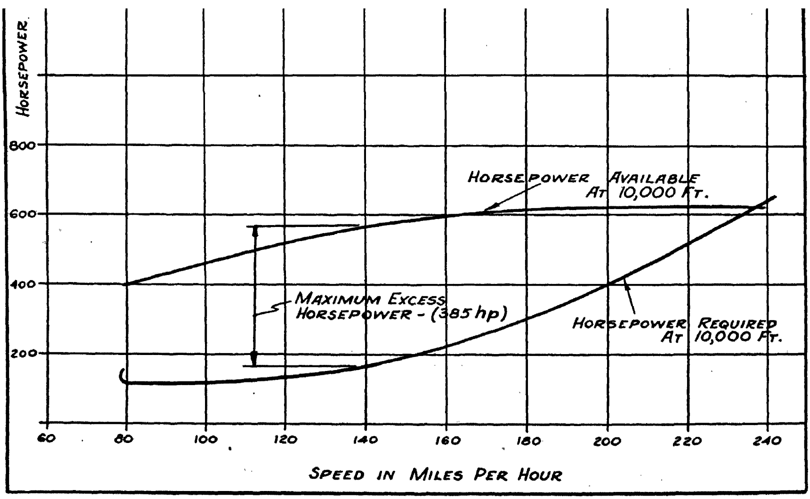


FIGURE 108

Rate of Climb

The rate of climb is calculated by means of the formula

$$\text{Rate of Climb in Feet per Minute} = \frac{33{,}000 \text{ (Excess Horsepower)}}{\text{Gross Weight in Pounds}}$$

The maximum difference between the curves of the horsepower available and the horsepower required at the same speed along the flight path will determine the maximum rate of climb at the particular altitude for which the calculations are made.

In Figure 103, the maximum excess power is 435 horsepower; the rate of climb is then

$$\frac{435 \times 33{,}000}{10{,}000} = 1436 \text{ feet per minute at}$$

sea level. Similarly the rates of climb at 7000 feet is 1419 feet per minute, and at 10,000 feet, the maximum rate of climb is 1270 feet per minute.

Absolute and Service Ceilings

The absolute ceiling is that altitude at which the rate of climb is zero feet per minute.

The service ceiling is that altitude at which the rate of climb is 100 feet per minute.

The absolute and service ceilings may be obtained graphically by determining the rates of climb for two different altitudes (outside of the supercharged region and extending the straight line drawn through these points to intersect the zero ordinate).

This has been done in Figure 109. The service and absolute ceilings, of course, also could have been calculated by means of trigonometric relationships for two different altitudes. Actually the rate of climb does not vary directly with the change in altitude but the assumption is sufficiently close for all practical purposes.

Range

The range may be calculated on the basis of the known horsepower, fuel consumption and speed. This method is likely to give too conservative value although more refined formulae are available. The method is known as Breguet's Formula.

$$\text{Range in miles} = 863 \left(\frac{L}{D}\right) \frac{e}{c} \log_{10} \left(\frac{W_0}{W_1}\right)$$

where L/D = the maximum overall effectiveness of the complete airplane
e = average propeller efficiency at cruising
c = average fuel consumption in pounds per brake horsepower per hour at cruising
W_0 = gross weight of airplane at start of flight
W_1 = W_0 less fuel weight.

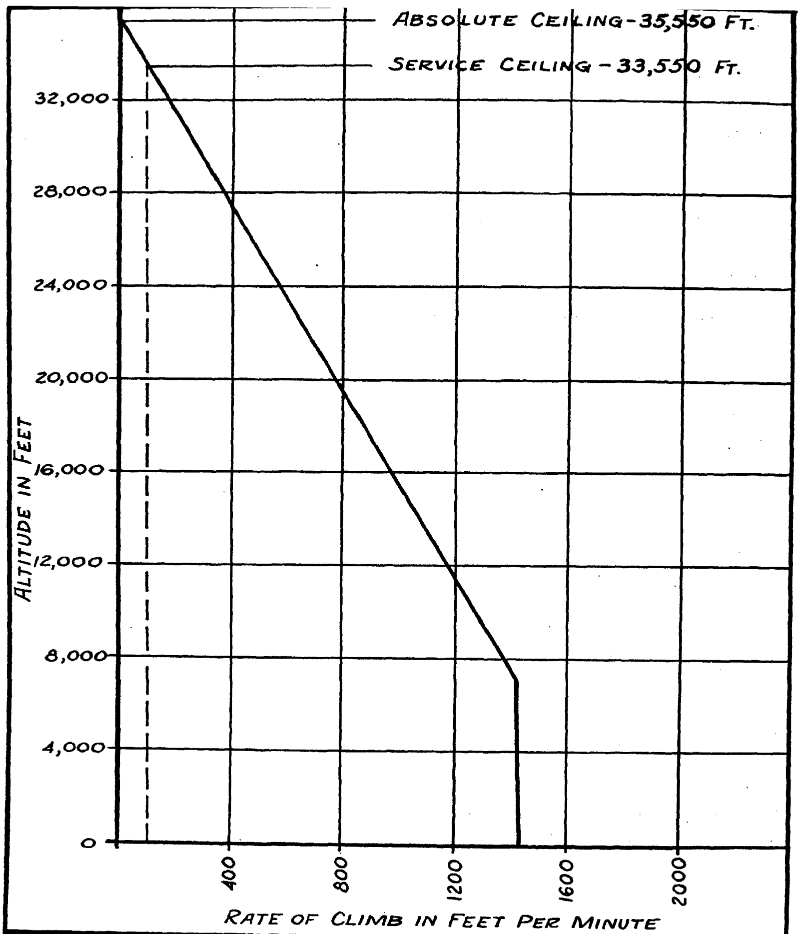


FIGURE 109

XVII. Airplane Data Sheets

The design of airplanes is largely empirical so that it is advisable to study as many airplanes as possible, catalog them under different categories, and collect as much data as possible on performance, weights, structure, power and all the other items that may be useful at sometime or another.

Table 37 below is what might be called a condensed airplane data sheet which may be taken as a guide as to the type of information that should be collected. It may be expanded to suit individual needs.

TABLE 37—*Condensed Airplane Data Sheets*

Name of Company ____________________

Name of Type ____________________

Price ____________________

1. *Power Plant*

   Engine ________ H.P. ________ R.P.M. ________

   Starter ____________________

   Design of Exhaust ____________________

   Other Engine Accessories ____________________

   Fuel and Oil Capacity ____________________

   Fuel and Oil Carried ____________________

   Disposition of Tanks ____________________

   Type of Engine Controls ____________________

   Propeller-Material, Diameter, Pitch ____________________

2. *Wing*	*Upper Wing*	*Lower Wing*
Airfoil Section	________	________
Wing Area	________	________
Total Wing Area (including ailerons)	________	________
Span	________	________
Chord—Root	________	________
Tip	________	________

Tip—Root ratio		
Aspect Ratio		
Mean Geometric Chord		
Dihedral		
Sweepback		
Incidence		
Length of Cantilever Tip		
Length of Outer Bay		
Length of Inner Bay		
Length of Center Section		
Location of Wing Spars in Per Cent of Chord		
Front:		
Rear:		
Maximum Rib Spacing		
Aileron Area		
Ratio of Aileron Area/Wing Area		
Flap Area		
Mean Gap		
Gap-chord Ratio		
Ratio of Gap to Mean Span		
Stagger in Degrees		
Efficiency of Upper Wing		
Location of Center of Gravity When Fully Loaded, in Per Cent of Mean Aero-Dynamic Chord		

3. *Tail Surfaces*

Stabilizer Area	
Elevator Area	
Total Horizontal Tail Surfaces	
Elevator/Horiz. Tail Surface	
Stabilizer/Horiz. Tail Surface	
Ratio of Horizontal Tail Surface Area to Wing Area	
Distance from Center of Gravity Loaded to Elevator Hinge	
Fin Area	

Rudder Area ____________

Total Vertical Tail Surface Area ____________

Fin/Vertical Tail Surface Area ____________

Rudder/Vertical Tail Surface Area ____________

Ratio of Vertical Surface Area to Wing Area ____________

Distance from Center of Gravity Loaded to Rudder Hinge ____________

Fin Area Forward of Center of Gravity Loaded ____________

Stabilizer Setting with Reference to Propeller Axis ____________

4. *Passenger Accommodations*

Dimensions of Cabin ____________

Finish ____________

Seating (type and weights, size, spacing) ____________

Aisle Width ____________

Aisle Height ____________

Windows (type) ____________

Doors and Steps (emergency, etc.) ____________

Heating ____________

Ventilation ____________

Lighting ____________

Baggage Accommodation ____________

Toilets ____________

5. *Pilot's Cockpit*

Single or Dual ____________

Releasable Controls ____________

Vision—Angles ____________

Windows and Their Construction ____________

Windshield ____________

Arrangement of Seats ____________

Are Seats Adjustable ____________

Are Rudder Bars Adjustable ____________

Instrument Board ____________

Door—(separate for pilot) ____________

6. *Auxiliary Equipment and Accessories*
   Night Flying—Radio ______
   Direction Finding ______
   Navigation Instruments, etc. ______
   Special Equipment ______

7. *Angles*
   Landing Angle of Airplane ______
   Angle in Side Elevation between Vertical through Axle and Line Connecting Center of Gravity and Axle ______
   Angle in Front Elevation between Vertical and Line Joining the Center of Gravity and the Point of Contact with the Ground at the Outer Wheel ______
   Angle between the Ground and a Line from the Point of Tangency of the Wheel with the Ground to the Wing Tip on the Same Side of the Plane of Symmetry of the Airplane As the Wheel ______
   Angle between the Ground and a Line Joining the Point of Contact with the Ground of the Deflected Tail Skid to the Tip of the Horizontal Tail Surfaces ______

8. *Performance*

	Sea Level	*Rated Altitude*
High Speed in M.P.H.		
Cruising Speed in M.P.H.		
Landing Speed in M.P.H.		
Climb at Sea Level in F.P.M.		
Climb to 10,000′ in Minutes		
Service Ceiling in Feet		
Fuel Capacity in Gallons		
Normal Range in Miles		
Normal Fuel Consumption at Cruising Speed Gal. per Hr.		

9. *Miscellaneous*

Distance from Front Face, Rear Propeller Flange to Center of Gravity of Airplane Loaded__________

Type of Fuselage Construction__________

Tread of Landing Gear__________

Type of Tail Skid Construction__________

Type of Chassis Construction__________

Design Load Factors:

High Incidence__________

Low Incidence__________

Inverted Flight__________

Landing__________

10. *Weights*

Weight Empty (lb.) with Water if Water-Cooled Engine Is Used__________

Payload (lb.)__________

Disposable Load (lb.)__________

Normal Gross Weight Loaded (lb.)__________

Wing Loading (lb. per sq. ft.)__________

Power Loading (lb. per hp.)__________

11. *Overall Dimensions*

Span__________

Length__________

Height__________

Tread__________

12. *Ratios*

Weight Empty/Gross Weight__________

Payload/Gross Weight__________

Disposable Load/Gross Weight__________

Wing Loading__________

Power Loading__________

Aileron Area/Wing Area__________

Vertical Tail Surface Area/Wing Area__________

Fin/Vertical Tail Surface Area__________

Horizontal Tail Surface Area/Wing Area__________

Elevator/Horizontal Tail Surface Area__________
Flap Area/Wing Area__________
Tab Area/Aileron Area__________
Tab Area/Rudder Area__________
Tab Area/Elevator Area__________
Tread/Span__________
Length/Span__________
Height/Span__________

13. *Three-View and Unusual Features*

Table 38 is a table of just such data as called for in Table 37. Some have been obtained from various magazines; others have been obtained by scaling three-view drawings, and in such cases the data given may not actually jibe with the actual design data.

TABLE 38—*Group 1, Small Commercial*

Reference						
Name	No. 1	No. 2	No. 3	No. 4	No. 5	No. 6
Type	H.W.M.E.	H.W.M.E.	L.W.M.C.	L.W.M.C.	L.W.M.C.	L.W.M.C.
Manufacturer						
Engine	WR.CY.SR. 1820 F42-G	WR.CY.	WR.CY. 1820F1-g	WR.CY.-F P.W.H.SDG	P.W.W.	WR.CY.-R 1820-F2
Rated Power	625	575	715	710	550	735
Rated R.P.M.	1950	1900	1950	1950	2200	1950
Power at Altitude	625			710cy	550	735
Gross Weight	6800	9690	7000	8750	5800	8500
Weight Empty	3850	5220	4100	5430	3325	5332
Weight Empty / Gross Weight	.567	.545	.586	.621	.574	.627
Useful Load	2950	4370	2900	3320	2160	3168
Useful Load / Gross Weight	0.434	0.456	0.414	0.379	0.373	0.373
Payload	2100	2880	2900	2200	Var.	1835
Payload / Gross Weight	0.309	0.301	0.415	0.252	Var.	0.216
Wing Area (S_w)	318.1	652	363	460	294.1	384
Wing Loading	18.8	14.7	19.2	19	19.72	20.
Gross Weight / Rated HP	11.3	16.6	9.8	12.5	10.54	10.15
Payload / Rated HP	3.36	5.0	4.06	3.1	Var.	2.5

Maximum Speed Sea Level		140	221	210	230	235
Cruising Speed Sea Level		122	201	190	205	215 10,000
Cruising Range	800 mi.	725	1700	680	720	1000
Landing Speed Sea Level		59	58	59	65	65
Rate of Climb Sea Level		700	1000	900	1500	1000
Service Ceiling		14000	24800	22000	22000	20000
Absolute Ceiling			26800		24500	22000
Airfoil Section at Wing Root			NACA 2400	Clark-Y	Clark-Y	
Airfoil Section at Wing Tip			NACA 2400	Clark-Y	Clark-Y	
Span (b)	60′	65′	48′	53′	42′9″	50′
Root Chord			114″	155″		135″
Tip Chord			64″	56.5″		60″
Mean Geometric Chord			99.5″	105″		100″
Tip / Root Ratio			.56	.364		.444
Aspect Ratio	7.86	6.48	6.34	6.10	6.22	6.51
Dihedral			4° top of wing	6°	4¼°	3° top of wing
Chord Incidence to Thrust Line			2½″		0°	
Aileron Chord			1.5′	18″		1.33′
Aileron Span			7′			15′
Aileron Area	43.4		21.0	30	22.4	39.8
Aileron Area / Wing Area	.137		.058	.065	.076	.059
Aileron Tab Area			None			None
Tab Area / Aileron Area			None			0
Flap Area			63			
Flap Span			36′ Total			
Flap Chord			1.75			
Landing Gear	SPL		CAN-SL	RET	RET	RET
Tread	109″	180″	108″	160″	139″	145″
Tread / Span	.182	.231	.188	.252	.271	.242

Length (L)	38′8″	42′9″	32′5″	42′8″	27′10″	37′0″
$\frac{\text{Length}}{\text{Span}}$	.773	.658	.677	.805	.650	.740
Landing Angle			11.5°	10.5°		10°
Angle A			21°	26°		12°
Angle B			40°	34°		42°
Angle C			18°	22°		18°
Center of Gravity Location	12″ above TL 30% back of LE of MAC	12″ above TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC
H.T.S. Area (S_t)	49.1		56.8	83.5	59.6	71.1
$\frac{St}{Sw}$	.154		.157	.182	.203	.185
Stabilizer Area	29.1		37.4	43.1	38.0	38.6
Elevator Area	20		19.4	40.4	21.6	32.5
Elevator Tab Area			None			2.4
$\frac{\text{Stabilizer Area}}{\text{H.T.S. Area}}$	.593		.658	.516	.638	.543
$\frac{\text{Elevator Area}}{\text{H.T.S. Area}}$	.407		.342	.484	.342	.457
$\frac{\text{Elev. Tab Area}}{\text{Elevator Area}}$			0			.074
L = Distance from C.G. to Elev. H.G. Line			21′	26′ 3½″		23.5
$\frac{\text{L}}{\text{Wing MAC}}$			2.53	3.0		2.82
$\frac{\text{St L}}{\text{Sw MAC}}$			.397	.289		.522
V.T.S. Area	29.2		20.0	48.5		24.5
$\frac{\text{V.T.S. Area}}{\text{Wing Area}}$	.092		.055	.105		.064
Fin Area	13.5		9.0	32	11	9.9
$\frac{\text{Fin Area}}{\text{V.T.S. Area}}$	.462		.450	.660		.405
Rudder Area	15.7		11	165		14.6
$\frac{\text{Rudder Area}}{\text{V.T.S. Area}}$	.538		.550	.340		.596
$\frac{\text{V.T.S. Area X L}}{S_w \times b}$			.025	.032		.03
Cabin L				16′ 4″		

Dimensions W			57″	53¾″		58
H			60″	61⅛″		68
Seating Capacity	8	15	9	12	7	8
Seating Pass. Arrangement			2 each side	5 each side		4 each side
Volume of Pass. Cabin				530 ft.^{3} incl. lav.		250 ft.^{3}
Baggage & Mail Arrangement			35 ft.^{3} aft.	52 wg. stub 35 un. ckpt.		60 ft.^{3}-25 aft. cabin
Baggage & Mail Capacity Lb.		750		300	152	450
Position of Spars in Cen. Sec.				Under cabin		Through cabin
Overall Height				150″		122″
R.P.M. at Altitude	1950 @ 7200			1950 @ 7000	2200 @ 5000	1950 @ 4000
$V_{max} = K\left(\frac{P}{S}\right)^{\frac{1}{3}}$ K =	—	146	177	181	186.5	189
$V_{max} = \frac{K V_{min} \sqrt[3]{\eta}}{\sqrt[3]{V_{min} \frac{W}{P}}}$ K =	—	23.4	31.2	32	31.1	31.5
$H = 40{,}000 \log_{10}\left(\frac{K}{\frac{W}{P}\sqrt{\frac{W}{S}}}\right)$ K =	—	68.8	50.5	61.6	53.2	50.2

Group 2 — Transport

Reference						
Name	No. 1	No. 2	No. 3	No. 4	No. 5	No. 6
Type	HWMC	Biplane	LWMC	LWMC	LWMC	LWMC
Manufacturer						
Engine	2 PWH	2 WR-CY	2 PWW 51-DI-G	2 PWH SDG	2 WRWW 420	3 Lycom. R-680-8A
Rated Power	1350 6000 Ft.	1400	1050	1400	840	720
Rated R.P.M.	2050	1950	2200	2150	2150	2000
Power at Altitude		1400	1050	1400		
Gross Weight	14000	16800	13650	17880	9400	8750
Weight Empty	8400	11235	8950	12000	5855	5670
$\frac{\text{Weight Empty}}{\text{Gross Weight}}$		.668	.655	.672	.624	.648
Useful Load	5600	5565	4700	5880	3545	3080
$\frac{\text{Useful Load}}{\text{Gross Weight}}$		.331	.345	.329	.376	.352
Payload	3400	3200	2572	3560	3545	1860
$\frac{\text{Payload}}{\text{Gross Weight}}$		.191	.188	.199	.377	.218
Wing Area (S_w)	686	1276	836.1	940	458.3	500
Wing Loading	20.4	13.15	16.30	19.20	20.45	17.5

$\frac{\text{Gross Weight}}{\text{Rated H.P.}}$	10.2	12.50	12.4	12.7	10.7	12.2
$\frac{\text{Payload}}{\text{Rated HP}}$		2.29	2.45	2.54	4.23	2.59
Maximum Speed Sea Level	225	170	182 @ 5000	210	215	
Cruising Speed Sea Level	205	150	165 @ 5000	195	190	140
Cruising Range	500	600	810	1225	800	370
Landing Speed Sea Level	65	59	58	61	65	
Rate of Climb Sea Level	1180	1000	830	1090	1350	1060 WHup 935 WHdn
Service Ceiling	19000	15500	18400	23200	18000	
Absolute Ceiling		17500	20500	24800		
Airfoil Section at Wing Root				NACA 2215	Clark-Y-18	
Airfoil Section at Wing Tip				NACA 2209	Clark-Y-9	
Span (b)	71′2.4″	82′	74′	85′	55′	60′
Root Chord		108″	180″	170″	145″	
Tip Chord		108″	90″	68″	48″	
Mean Geometric Chord		108″	140″	126.5″	105″	
Aspect Ratio	7.36	5.27	6.55	7.70	6.60	7.20
$\frac{\text{Tip}}{\text{Root Chord}}$		1	.5	.4	.331	
Dihedral				5° median line	5° 34′ median line	
Chord Incidence to Thrust Line				2°	0°	
Aileron Chord		2	1.84	2.26	1.49	1.61
Aileron Span		26	17	19.2	10.67	16.6
Aileron Area		104	62.6	86.8	29.2	53.5
$\frac{\text{Aileron Area}}{\text{Wing Area}}$		.082	.075	.092	.064	.107
Aileron Tab Area		None	2.4	1.56	None	None
$\frac{\text{Tab Area}}{\text{Aileron Area}}$		0	.038	.018	0	0
Flap Area				117.8	44.4	41.6
Flap Span				41′	29.84	21.1
Flap Chord				2.77	1.49	1.98
Landing Gear	Ret.	Ret.	Ret.	Ret.	Ret.	Ret.

Tread		243	207	216	163	208
Tread / Span		.247	.233	.212	.246	.289
Length (L)	42′ 8″	48-7	51-4	62-0	38-7	36-10
Length / Span	.60	.593	.693	.730	.701	.613
Landing Angle		11°	8.5°	12.5°	13.3°	11.5°
Angle A		24°	16°	11°	12.6°	
Angle B		40°	42°	45°	42.7°	53°
Angle C		15°	16°	17°	23.5°	18.5°
Center of Gravity Location		on TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC
H.T.S. Area (S_t)		183.8	130.5	150.6	81.6	105
$\frac{S_t}{S_w}$		.144	.156	.160	.179	.210
Stabilizer Area		101.2	82.8	92.2	48.0	56.2
Elevator Area		82.6	47.7	58.4	33.6	48.8
Elevator Tab Area		4.5	3	8.65	2.39	1.9
Stabilizer Area / H.T.S. Area		.550	.635	.613	.588	.534
Elevator Area / H.T.S. Area		.450	.365	.387	.412	.466
Elev. Tab. Area / Elevator Area		.054	.063	.148	.071	.039
L = Distance from C.G. to Elev. H.G. Line		29	31	35½	25.2	
L / Wing MAC.		3.11	2.66	3.36	2.92	
$\frac{S_t\ L}{S_w\ MAC.}$		.464	.415	.209	.513	
V.T.S. Area		71.0	41.7	64.6	31.9	35
V.T.S. Area / Wing Area		.056	.050	.069	.078	.070
Fin Area		28.5	17.6	22.1	15.0	15
Fin Area / V.T.S. Area		.402	.422	.342	.471	.428
Rudder Area		42.5	24.1	42.5	16.1	20
Rudder Area / V.T.S. Area		.573	.578	.658	.529	.572
$\frac{\text{V.T.S. Area} \times L}{S_w \times b}$		.0197	.021	.029	.032	

Cabin Dimensions L						
W			20′	26′ 4″		
H		6′ 3″	6	5′ 6″	58½″	
Seating Capacity	16	18	12	6′ 3″	60″	
Seating Pass. Arrangement	3 double seats ea. side	3 abreast	2 abreast	16 7 each side	12 5 each side	10 Ham. type 4 facing center
Volume of Pass. Cabin		856		858 ft.³	356 ft.³	300
Baggage & Mail Capacity Lb.		650	300	480	500	500
Baggage & Mail Arrange.				76 ft.²F 112 ft.³R	54 ft.³ nose W. stub	R. of cab. Nacelles
Position of Spars in Cen. Sec.				Under cabin floor	Through cabin aft. of 2 front seats	
Overall Height	10′	193″		192″	121″	170″ inc. mast
R.P.M. at Altitude		1950 @ 7000	2200 @ 5000	2150 @ 6000		
$V_{max} = K\left(\frac{P}{S}\right)^{\frac{1}{3}}$ K	179.5	157.5	170	192	176	
$V_{max} = \underline{KV_{min}\sqrt[3]{\eta}}$	30.1	25.9	28.2	31.5	29.4	
$H = 40{,}000 \log_{10}\left(\frac{K}{\frac{W}{P}\sqrt{\frac{W}{S}}}\right)$	51.8	49.6	54.7	61.8	58.6	

Group 3 — Private Type

Reference				
Name	No. 1	No. 2	No. 3	No. 4
Type	Biplane	Biplane	HWMC	HWMC
Engine	JA	JA	WAR	LY
Rated Power	225	225	145	245
Rated R.P.M.	2000	2000	2050	2300
Power at Altitude				
Gross Weight	3150	3100	2200	3550
Weight Empty	1800	1900	1220	2310
$\frac{\text{Weight Empty}}{\text{Gross Weight}}$	.572	.612		
Useful Load	1350	1210	980	1240
$\frac{\text{Useful Load}}{\text{Gross Weight}}$	.43	.39		
Payload		700		
$\frac{\text{Payload}}{\text{Gross Weight}}$		.226		

Wing Area (S_w)	273	238.8	180.5	230
Wing Loading	11.5	12.98	12.3	15.4
$\frac{\text{Gross Weight}}{\text{Rated HP}}$	14	13.78	15.3	14.5
Maximum Speed Sea Level	175	155	162	133
$\frac{\text{Cruising Speed}}{\text{Sea Level}}$	155	137	143	120
Cruising Range	550-750		550	
Landing Speed Sea Level	45	53	47	55
Rate of Climb Sea Level	1000	1000	1000	850
Service Ceiling	15000	16300	18900	14500
Airfoil Section at Wing Root	Clark Y-H			
Span (b)	32′	35′	33′ 10″	41′
Root Chord	5′			
Tip Chord	5′			
Mean Geometric Chord	5′			
$\frac{\text{Tip}}{\text{Root}}$ Ratio	1			
Aspect Ratio	6.4	5.12	6.4	7.31
Aileron Area	23.8			28.2
$\frac{\text{Aileron Area}}{\text{Wing Area}}$	.142			
Aileron Tab Area	None	None	None	None
Flap Area	6	None		
Flap Span	12′ Total			
Flap Chord	6″			
Landing Gear	Ret.	Non-ret.	Can	Non-ret.
Tread	86″	86″		118″
$\frac{\text{Tread}}{\text{Span}}$	.218	.2		.24
Length (L)	24′ 5″	25′ 4″	24′ 7″	27′ 2″
$\frac{\text{Length}}{\text{Span}}$	.765	.724	.725	.66
Landing Angle	14°	17°	15°	14°
Angle A	18°		16°	20°
Angle B	35°		32°	42°
Angle C	17°		27°	30°

Center of Gravity Location	30% MAC	30% MAC on TL	30% MAC 12″ above TL	30% MAC 12″ above TL
H.T.S. Area (S_t)	38.9			43.4
$\frac{S_t}{S_w}$	.143			
Stabilizer Area	23.8			26.3
Elevator Area	15.1			17.1
Elevator Tab Area	.78			None
$\frac{\text{Stabilizer Area}}{\text{H.T.S. Area}}$	.261			.606
$\frac{\text{Elevator Area}}{\text{H.T.S. Area}}$	.388			.395
$\frac{\text{Elev. Tab. Area}}{\text{Elevator Area}}$	.051			
V.T.S. Area	14.2			24.3
$\frac{\text{V.T.S. Area}}{\text{Wing Area}}$	.052			.106
Fin Area	5.5			10.6
$\frac{\text{Fin Area}}{\text{V.T.S. Area}}$	.387			.436
Rudder Area	8.7			13.7
$\frac{\text{Rudder Area}}{\text{V.T.S. Area}}$	.612			.564
$\frac{\text{V.T.S. Area} \times L}{S_w \times b}$	.0255			
Cabin Dimensions L / W / H	76 / 40″ / 45″			
Seating Capacity	4-5	4	4	4
Seating Pass. Arrangement	2 each side	2 each side	2 each side	2 each side
Volume of Pass. Cabin	79.5 Ft.^{3}			
Baggage & Mail Arrangement	Rear of cabin		Rear of cabin	
Position of Spars in Cen. Sec.	None		Overhead	
Overall Height	8′ 6″	8′ 1″		8′ 6″
$V_{max} = K\left(\frac{\rho}{s}\right)^{\frac{1}{3}}$ K $\doteq$	187	159	174	130
$V_{max} = \frac{K V_{min} \sqrt[3]{\eta}}{\sqrt[3]{V_{min}\frac{W}{P}}}$ K =	33.4	26.74	30.8	22.4
[illegible] $\left(\frac{K}{\ }\right)$ K =	51.8	54	52.8	61

Group 4 — Foreign Transport

Reference							
Name	No. 1	No. 2	No. 3	No. 4	No. 5	No. 6	No. 7
Type	Biplane	Biplane	LWMC	LWMC	LWMC	HWMC	HWMC
Manufacturer							
Engine	4 Bristol Jupiter X FBM	4 DH Gipsy 6	BMW VI	3 HIS. Suiza 9V-Rad.	3 Gnome RH 7KD	3 WR-Cy R-1820-F	2 Armsg. Siddeley Jaguar
Rated Power			630	1725	1050	1920	920
Rated R.P.M.			1600	1900		1900	2000
Gross Weight	33500	9200	7282	20591	13640	19505	11790
Weight Empty	22650	5566	5060	11618	8334	11790	7360
Weight Empty / Gross Weight	.676	.605	.695	.565	.61	.603	.625
Useful Load	10850	3634	2222	8973	5306	7715	4430
Useful Load / Gross Weight	.324	.395	.306	.436	.39	.395	.377
Payload		1915	1100	2646	2720	4235	2460
Payload / Gross Weight		.208	.151	.129	.199	.217	.251
Wing Area	2615	600	393	1033	686.5	1033 ft.^{2}	728
Wing Loading	12.81	15.35	18.55	19.9	19.9	18.9	16.2
Gross Weight / Rated HP	11.2	11.55	11.9		12.9	10.0	12.8
Payload / Rated HP	2.4	1.75	1.54		2.59	2.21	3.22
Max. Speed Sea Level		170	226	186	156.4	186	160
Cruis. Speed Sea Level		145	201	155		154	135
Cruis. Range			620	1243		1000	600
Landing Speed Sea Level		66	68.4	62		59	64
Rate of Climb Sea Level			940	870 @ 0-9800 ft.	615	910 0-3280′	970
Service Ceiling		19000	17600	19685	17056	17500	15500
Absolute Ceiling				21325		19500	
Airfoil Sec. at Wing Root		Mod Raf. 34					
Airfoil Sec. at Wing Tip		Mod Raf. 34					
Span (b)	Up 113-0 LW 92-6	64-6	48.5	95.14	71′ 1″	84.2	71′ 3″

Root Chord	13-9	84	124.0	14.76		192.	151.5″
Tip Chord	13-9		Elliptical Planform	7.38′		111″	96.5″
Mean Geometric Chord	13-9		110.5″	11.5′		155.5″	127.6″
Tip / Root Ratio	1.00		—	.5		.578	.637
Aspect Ratio	8.78		5.95	8.8		8.11	6.96
Dihedral			6° median line	5° median line			1.5°
Chord Incid. to Thrust Line							0°
Aileron Chord			1.22	2		1.68	1.2
Aileron Span			11.75	41.5		16	18.5
Aileron Area	173		28.7	162		54	44.6
Aileron Area / Wing Area	.066		.073	.157		.052	.061
Aileron Tab Area	None	None	None	None		None	None
Tab Area / Aileron Area	0	0	0	0		0	0
Flap Area		None	None	None	None	54 total	None
Flap Span		0	0	0		½ semi wg. span	0
Flap Chord		0	0	0		1.68	0
Landing Gear	Non-ret.	Can-sl.	Ret.	Can-sl.		Ret.	Non-ret.
Tread	23′ 9″	155	118″	21.65′		236	190
Tread / Span	.198	.201	.203	.228		.215	.222
Length (L)	83′ 10″	45-11	37.7′	62.17	55′ 9″	54′ 9″	54-6
Length / Span	.698	.628	.778	.654	.753	.591	.767
Landing Angle	13.5°	13½°	12°	8½°		11°	7°
Angle A	7°	9°	14°	15°		11°	11°
Angle B	33.7°	41°	43°	62°		46°	43°
Angle C	24°	15°	21.5°	17°		25°	20°

Center of Gravity Location	12″ below TL 30% back of LE of MAC	12″ above TL 30% back of LE of MAC	12″ below TL 30% back of LE of MAC	35% MAC 12″ below TL	12″ below TL 30% back of LE of MAC	12″ above TL 30% back of LE of MAC	12″ above TL 30% back of LE of MAC
H.T.S. Area (S_t)	334	57.5	44			104.3	87.5
$\frac{S_t}{S_w}$	.128	.096	.112			.101	.120
Stabilizer Area	206	29.9	27.6			65.6	47
Elevator Area	128	27.6	16.4			38.7	40.5
Elevator Tab Area	None	None	None	None		None	None
Stabilizer Area / H.T.S. Area	.617	.52	.627			.629	.537
Elevator Area / H.T.S. Area	.383	.43	.373			.371	.463
Elev. Tab Area / Elevator Area	0	0	0			0	0
L = Distance from C.G. to Elev. H.G. Line	41.75	28.5	24-4			36.2	33.25
L / Wing M.A.C.	3.03	5.4	2.62			2.80	3.13
$\frac{S_t \ L}{S_w \ M.A.C.}$	0.388	.516	0.293			.283	0.376
V.T.S. Area	152	31.0	16.55			58.2	38.9
V.T.S. Area / Wing Area	.058	.052	.042			.056	.053
Fin Area	79	11.1	11.05			31.6	14.1
Fin Area / V.T.S. Area	.520	.358	0.668			.541	.362
Rudder Area	73	19.9	5.5			26.0	24.8
Rudder Area / V.T.S. Area	.480	.642	.332			.459	.638
$\frac{\text{V.T.S. Area} \times L}{S_w \times b}$	.02	.023	.021			.024	.025
Cabin Dimensions L	*Fwd* 6-7 | *Rr* 21-10	21-0		21-4		16-0	21-0
W	10-4 | 10-9	5-0		5-7		5-0	5-0
H	7-0 | 7-4	6-0		6-7		6-0	6-0
Seating Cap.	40	12		11		15	18
Seating Pass. Arrangement				4 e. side reclining		6 each side	8 each side
Volume of Pass. Cabin	2185	508		780		425	630

Baggage & Mail Arrangement	58 8 C.F. aft. cabin	42 C.F. aft. cabin		25 C.F. aft. cabin		235 ft.^{3} fuse 4 WG	
Baggage & Mail Capacity Lb.		400		400 (variable)			450
Position of Spars in Cen. Sec.				Thru cab. bet. pass. & pilot			Through cabin
Overall Hght.	29′ 6″	126″	122″	17′ 8″		157	138″
R.P.M. at Altitude	—	—	—	—	—	—	—
$V_{max} = K\left(\frac{P}{S}\right)^{\frac{1}{3}}$ K =	—	140	193	—	135	151	157.5
$V_{max} = \frac{K V_{min} \sqrt[3]{\eta}}{\sqrt[3]{V_{min} \frac{W}{P}}}$ K =	—	25.3	30.4	—	—	26.5	24.8
$H = 40{,}000 \log_{10}\left(\frac{K}{\frac{W}{P}\sqrt{\frac{W}{S}}}\right)$ K =	—	49	55.5	52.8	62	48.2	56.3

Group 5 — Miscellaneous

Reference			
Name			
Type	Biplane	LWM	HWMC
Manufacturer	No. 1	No. 2	No. 3
Engine	PWW	PWW	WR-WW
Rated Power	450 5000′	550 5000′	175
Rated R.P.M.	2000	2200	2000
Power at Altitude			
Gross Weight	3200	5800	2780
Weight Empty	2120	3550	1720
Useful Load	1080	2250	1060
Wing Area (S_t)	216	294	208
Wing Loading	14.8	19.72	13.3
$\frac{\text{Gross Weight}}{\text{Rated HP}}$	7.5	10.54	15.8
Maximum Speed Sea Level	5000′ 190	5000′ 220	142
Cruising Speed Sea Level	5000′ 155	11000′ 205	121
Landing Speed Sea Level	60	63	47
Rate of Climb Sea Level	2500	1400	700

Service Ceiling	25000	24000	17000
Span (b)	30′ 6″	42′ 9″	36′
Aileron Area	—	22.4	—
Landing Gear	Non-ret.	Ret.	Non-ret.
Tread	—	—	—
$\frac{\text{Tread}}{\text{Span}}$	—	—	—
Length (L)	22′ 8″	28′ 4″	26′ 10″
$\frac{\text{Length}}{\text{Span}}$	.745	.66	.748
Landing Angle	—	15°	—
Angle A	—	22°	—
Angle B	—	45°	—
Angle C	—	22°	—
Center of gravity location	30% MAC on TL	30% MAC 12″ below TL	30% MAC 12″ above TL
H.T.S. Area (S_t)	26.0	59.6	—
$\frac{S_t}{S_w}$			
Stabilizer Area	—	38	—
Elevator Area	—	21.6	—
Elevator Tab Area	None	—	None
$\frac{\text{Stabilizer Area}}{\text{H.T.S. Area}}$	—	.637	—
$\frac{\text{Elevator Area}}{\text{H.T.S. Area}}$			
Elev. Tab. Area	—	.362	—
Elev. Area	0	—	0
V.T.S. Area	13.0	23.8	
Fin Area		11	—
Rudder Area		12.8	
$\frac{\text{Rudder Area}}{\text{V.T.S. Area}}$		.538	
Seating Capacity	2	1 2	4

Seating Pass. Arrangement	Tandem	Tandem	2 each side
Volume of Pass. Cabin	None	None	
Overall Height	9′	9′ 3″	
$V_{max} = K\left(\frac{P}{S}\right)^{\frac{1}{3}}$ K	167	178	155
$V_{max} = \frac{K V_{min} \sqrt[3]{\eta}}{\sqrt[3]{V_{min} \frac{W}{P}}}$	24.2	30.5	27.2
$H = 40{,}000 \log_{10}\left(\frac{K}{\frac{W}{P}\sqrt{\frac{W}{S}}}\right)^{1}$	33.3	54	61

Angle A —Angle between vertical thru axle & line joining CG & axle.

Angle B —Angle in front elevation between vertical & line joining CG to pt. of contact with ground at outer wheel.

Angle C —Angle between ground & line from wing tip to pt. of tangency of wheel to ground.

L — Distance of CG to elevator hinge line.

b —Wing span.

S —Area, sq. ft.

A breakdown of weights of various airplanes is very useful when preparing weight estimates. Table 39 indicates how the weight breakdown may be made.

TABLE 39 — *Weight Breakdown*

Name of Company

Name of Type

Price

Power Plant

Engine

Per Cent of Power Plant
Per Cent of Weight Empty
Per Cent of Gross Weight
Cowling
Propeller
Radiators
Radiator Piping
Expansion Tank (water)
Oil Radiator
Fuel Tanks
Fuel Piping
Engine Water

Radiator Water
Oil Tank
Oil Piping
Engine Controls
Exhaust Manifold
Total Power Plant
Per Cent of Weight Empty
Per Cent of Gross Weight

Combustible Load

Gasoline
Oil
Total
Per Cent of Useful Load
Per Cent of Gross Weight

Crew

Pilot (s)
Mechanic (s)
Stewardess
Radio Man
Navigator
Others
Total
Per Cent of Useful Load
Per Cent of Gross Weight

Payload

Passengers
Baggage
Freight
Mail, Parcel Post
Miscellaneous
Total Payload
Per Cent of Useful Load
Per Cent of Gross Weight

Furnishings

Flooring
Fire Wall
Surface Controls
Instrument Board
Control Wires
Seats

Cushions
Soundproofing
Heating and Ventilating
Toilets
Miscellaneous
Total Furnishings
Per Cent of Weight Empty
Per Cent of Gross Weight

Equipment
Starting System
Radio Equipment
Photo Equipment
Parachute
Fire Extinguishers
Oxygen Apparatus
First Aid
Lighting
Total Equipment
Per Cent of Weight Empty
Per Cent of Gross Weight

Wing Group
Planes, Upper Lower
Ailerons
Struts
Wires
Flaps
Tabs
Hydraulic Mechanisms
Total Wing Group
Pounds per Square Foot
Per Cent of Structure
Per Cent of Weight Empty
Per Cent of Gross Weight

Body Group
Fuselage, Including Covering
Engine Nacelles
Cowlings
Total Body Group
Per Cent of Structure
Per Cent of Weight Empty
Per Cent of Gross Weight

Empennage
- Elevators
- Rudder
- Stabilizer
- Fin
- Tabs
- Struts
- Wires
- Total Tail Group
  - Per Cent of Structure
  - Per Cent of Weight Empty
  - Per Cent of Gross Weight

Landing Gear Group
- Chassis
- Shock Absorbers
- Wheels
- Cowling
- Tail Wheel
- Retracting Mechanisms
- Total Landing Gear Group
- Per Cent of Structure
- Per Cent of Weight Empty
- Per Cent of Gross Weight

Total

The following abbreviations and definitions are useful

W.E. = weight empty = structure plus power plant plus furnishings.
U.L. = useful load = crew plus combustible load plus pay load.
C.L. = combustible load = gasoline plus oil.
G.W. = gross weight = weight empty plus useful load.
D.L. = disposable load = crew plus easily removable equipment and furnishings plus combustible load.

APPENDIX

COMMUNICATION SYSTEMS

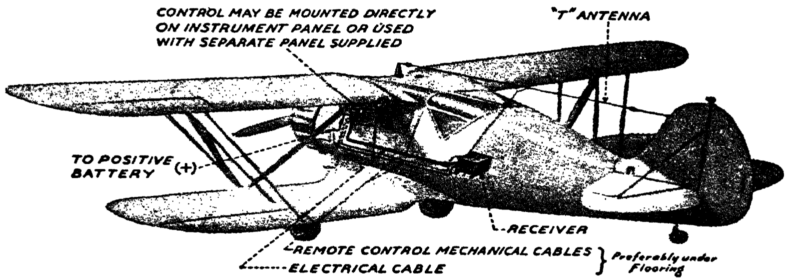


FIGURE 110 A

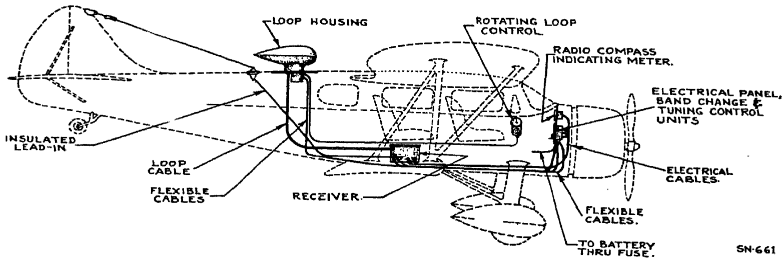
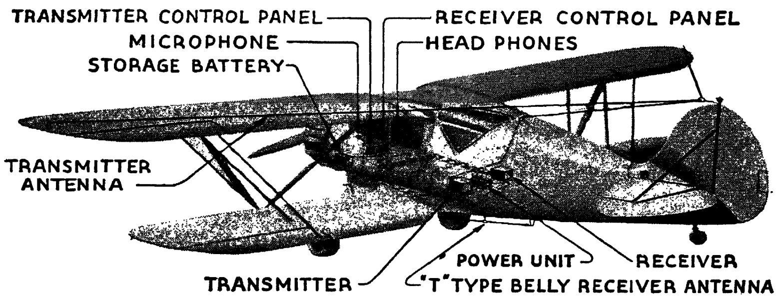


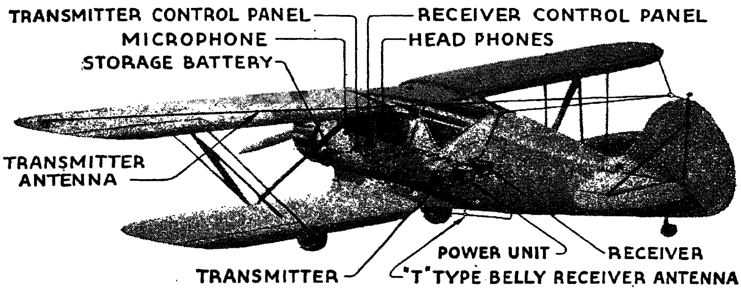
FIGURE 110 B TYPICAL INSTALLATION OF RADIO EQUIPMENT IN A SINGLE ENGINE AIRPLANE



Note: Free Trailing Non-retractable, Weightless Antenna can be used with much greater Efficiency in Place of Fixed Antenna Shown.

INSTALLATION of AVT-12 or 12A TRANSMITTER and AVR-7, 7A, 7B or 7C RECEIVER

FIGURE 111 A. RADIO EQUIPMENT MANUFACTURED BY R. C. A.



Note: Free Trailing Non-retractable, Weightless Antenna can be used with much greater Efficiency in Place of Fixed Antenna Shown.

INSTALLATION of AVT-7 or 7A TRANSMITTER and AVR 7 or 7A RECEIVER

FIGURE 111 B

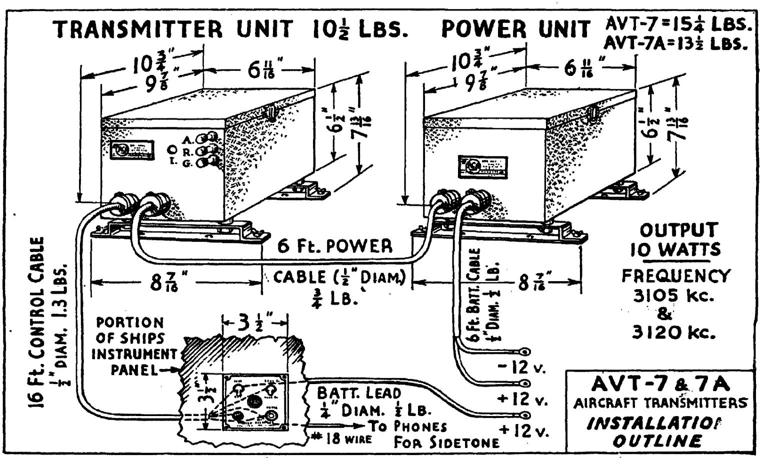


FIGURE 111 C. RADIO EQUIPMENT MANUFACTURED BY R. C. A.

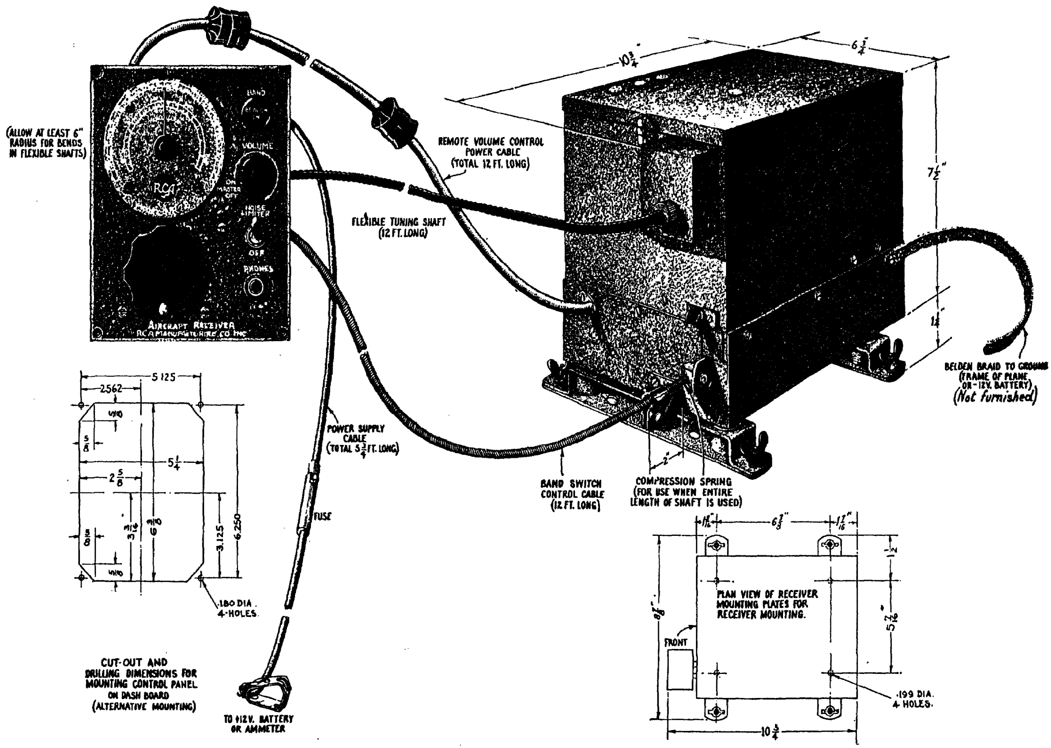


FIGURE 111 D. MOUNTING DETAILS OF AVR-7-B AND AVR-7-C RECEIVERS

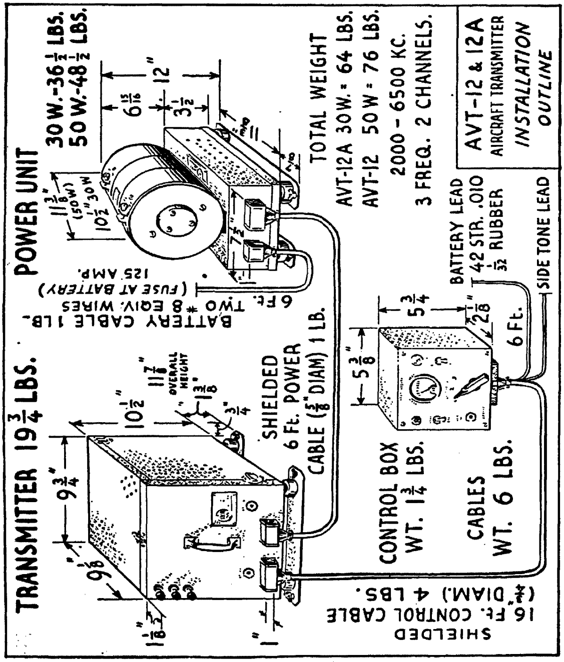

TRANSMITTER 19¾ LBS.
POWER UNIT
30 W.-36½ LBS.
50 W.-48½ LBS.
BATTERY CABLE 1 LB.
6 Ft. TWO #8 EQIV. WIRES
(FUSE AT BATTERY)
125 AMP.
SHIELDED
6 Ft. POWER
CABLE (5/8" DIAM) 1 LB.
TOTAL WEIGHT
AVT-12A 30 W. = 64 LBS.
AVT-12 50 W = 76 LBS.
2000 - 6500 KC.
3 FREQ. 2 CHANNELS.
AVT-12 & 12A
AIRCRAFT TRANSMITTER
INSTALLATION
OUTLINE
BATTERY LEAD
42 STR. .010
1/32 RUBBER
SIDE TONE LEAD
6 Ft.
CONTROL BOX
WT. 1¼ LBS.
CABLES
WT. 6 LBS.
SHIELDED
16 Ft. CONTROL CABLE
(¾" DIAM.) 4 LBS.

FIGURE 111 E

TABLE 40

Types	Voltage	Data on Types AC, TS, TSN, and Tx Exide Batteries									
		Dimensions in Inches			Net Weight in Lb.	5-Hr. Amp. Hr.	20-Min. Amps.	10-Min. Amps.	5-Min. Amps.	75 Amp. Min. to 10 Volts	Application See Below
		L.	W.	H.							
6-TX-9-1	12	7-3/16	7-1/4	10-13/16	38	29	50	84	133	10.0	1
6-TX-13-9	12	9-13/16	7-1/4	10-13/16	53	43	75	126	200	19.0	1
6-TX-19-1	12	13-15/16	7-1/4	10-13/16	70	65	112	189	300	33.0	1
3-TX-19-1	6	7-15/16	7-5/16	11-5/16	35	65	112	189	300	....	2
1-TX-19	2	3	7-1/4	10-3/4	14	65	112	189	300	....	3
6-TS-7-1	12	7-11/16	5-3/16	11-9/32	26	19	30	48	75	3.5	2
6-TS-13-1	12	10-5/16	5-3/16	11-9/32	36	38	60	96	150	14.0	1
6-TSN-25-3	12	15	6-9/16	11-3/8	67	75	120	189	300	36.5	1
3-TS-13-1	6	5-13/16	5-5/16	11-9/32	18	38	60	96	150	....	2
6-AC-7-1	12	8-1/16	3-7/8	6-13/16	17	10	15	24	36	....	4
4-AC-7-1	8	5-7/16	3-7/8	6-13/16	11	10	15	24	36	....	5
1-AC-11	2	4-5/16	2-7/8	8-3/8	5	17	25	39	60	....	3

Application

1. Cranking, lighting, radio and ignition. For use with high wattage landing lamps.
2. Cranking, limited lighting, radio, ignition.
3. Radio or other low voltage requirements.
4. Ignition.
5. Limited lighting and ignition.

FLIGHT INSTRUMENTS

FIGURE 112. USUAL ARRANGEMENT OF AIRSPEED, BANK AND TURN, AND CLIMB INDICATORS. *Pioneer* INSTRUMENTS SHOWN

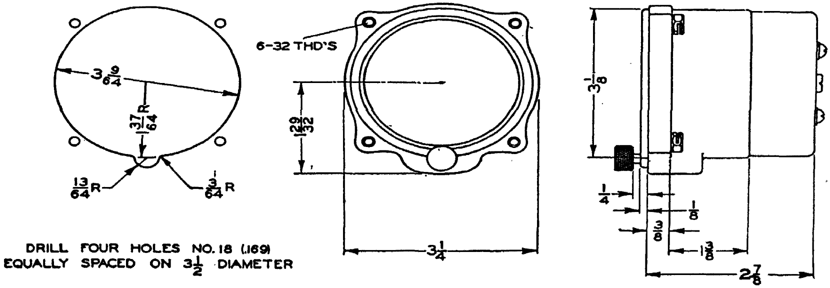


FIGURE 113. KOLLSMAN ACCELEROMETER

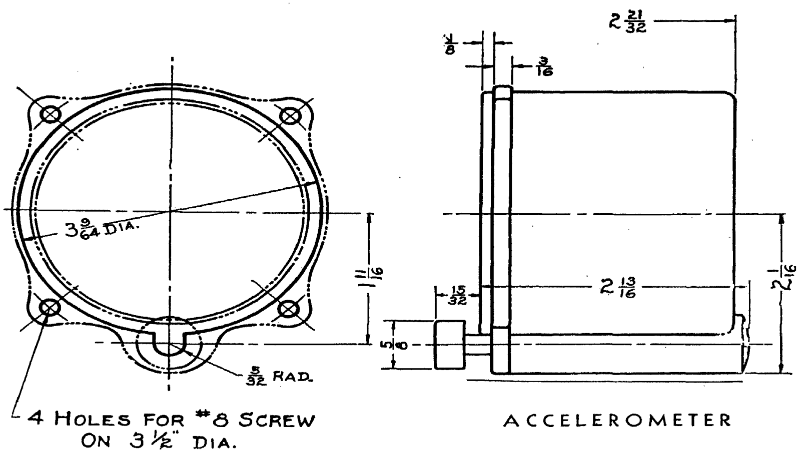


FIGURE 114. PIONEER ACCELEROMETER

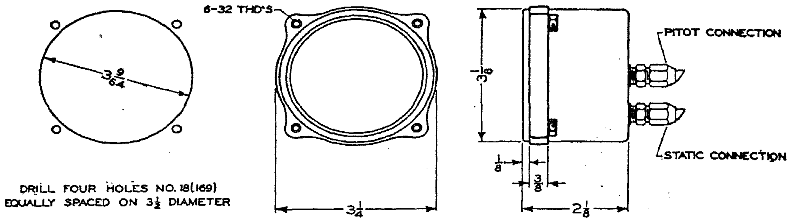


FIGURE 115. KOLLSMAN AIRSPEED INDICATOR

Ranges: 0-220 m.p.h.
0-250
0-300
0-350
0-300-500

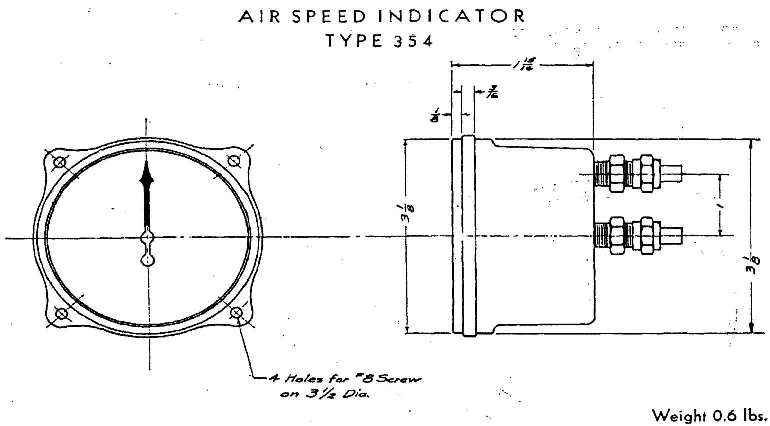


FIGURE 116. PIONEER AIRSPEED INDICATOR

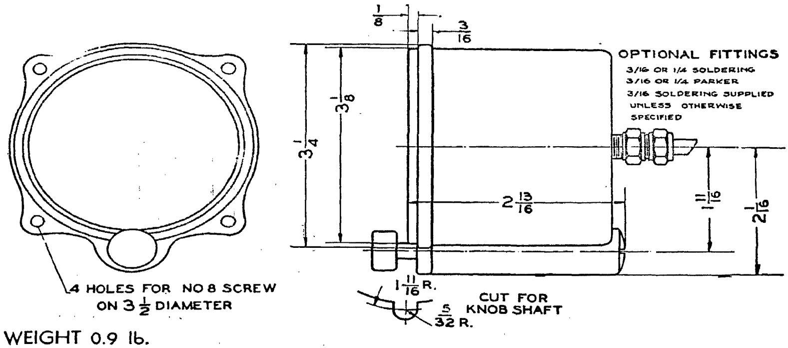
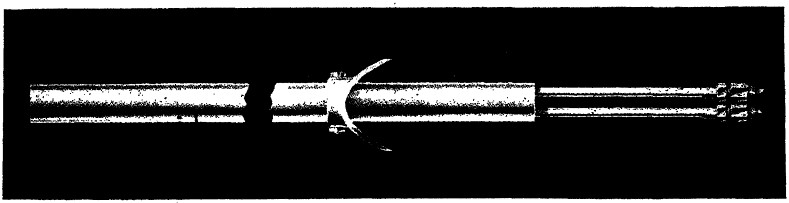
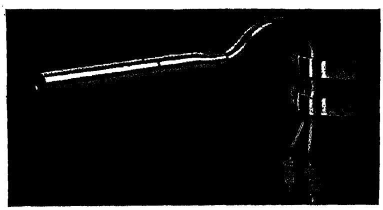


FIGURE 117. PIONEER AIRSPEED INDICATOR



TYPE 112



TYPE 10

FIGURE 118. KOLLSMAN PITOT-STATIC TUBES FOR AIRSPEED INDICATORS

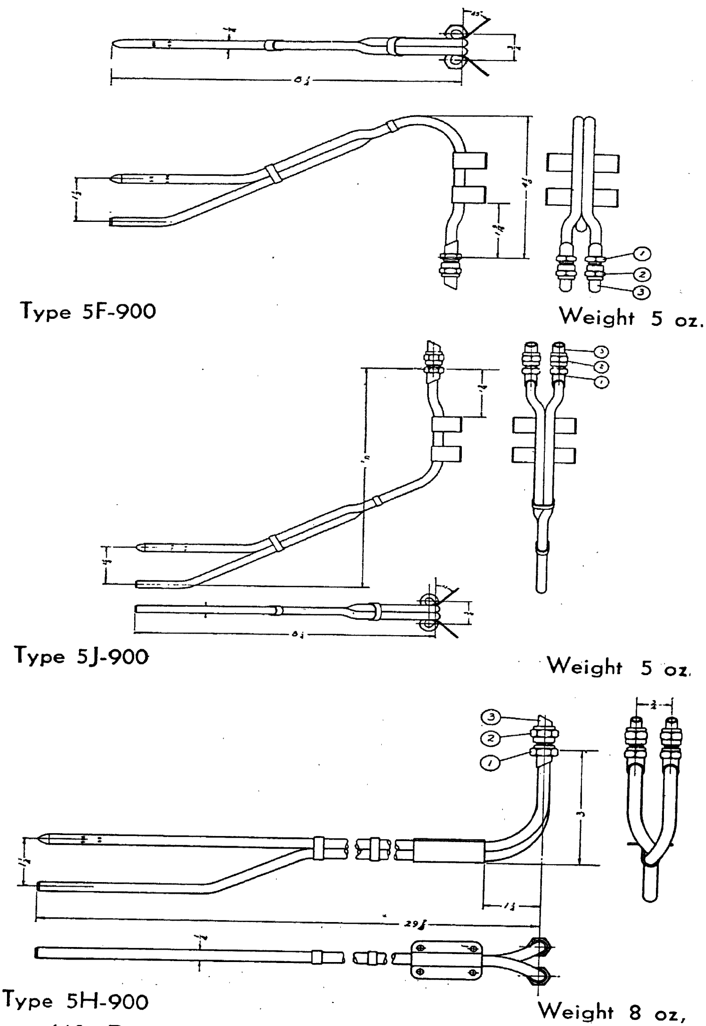
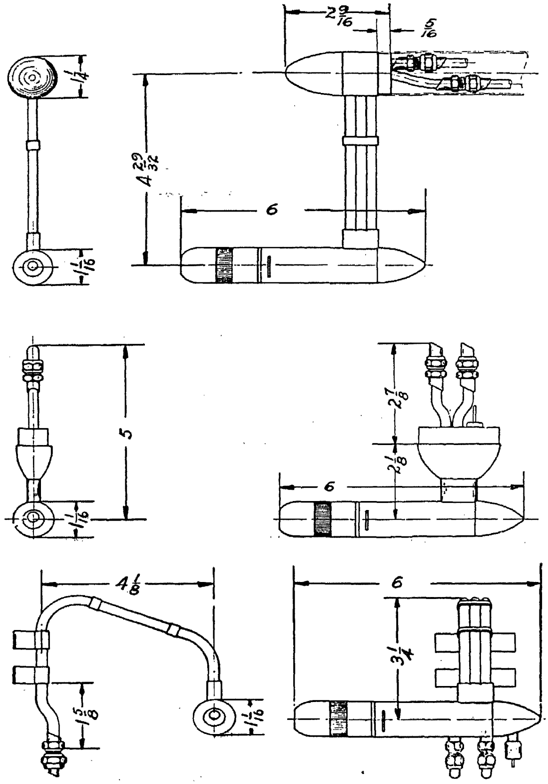


FIGURE 119. PIONEER PITOT-STATIC TUBES FOR AIRSPEED INDICATORS



3/16 AN fittings supplied unless otherwise specified. 1/4 AN or either size Parker fitting furnished optionally.

TYPE 357 F-940

Weight 0.9 lbs.

TYPE 357 F-914

Weight 0.9 lbs.

TYPE 357 F-905

Weight 0.7 lbs.

FIGURE 120. ELECTRICALLY HEATED PITOT-STATIC TUBES FOR AIRSPEED INDICATORS

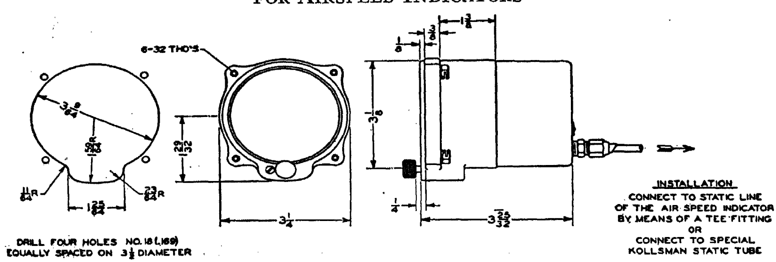


FIGURE 121. KOLLSMAN SENSITIVE ALTIMETER

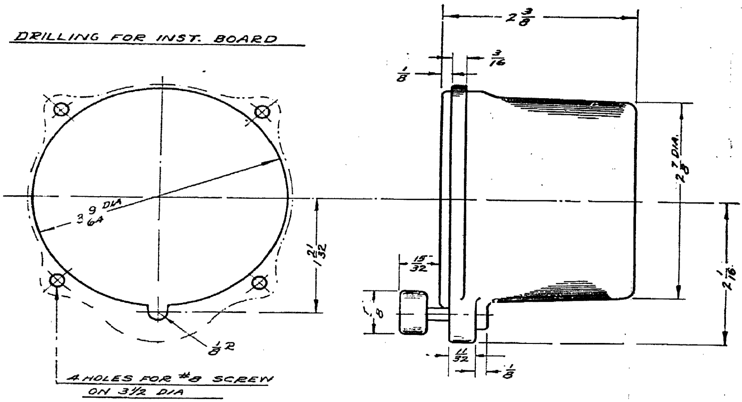
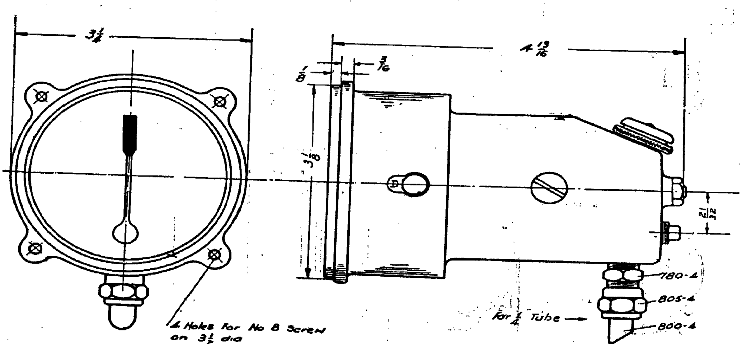


FIGURE 122. PIONEER ALTIMETER



WEIGHT 1.2 lb.

FIGURE 123. PIONEER BANK AND TURN INDICATOR

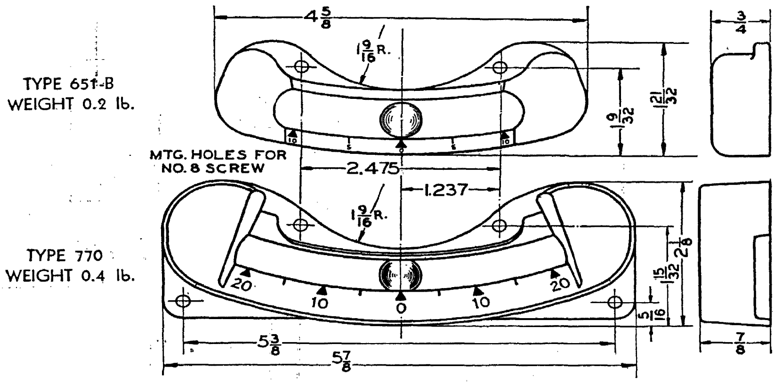


FIGURE 124. PIONEER BANK INDICATOR

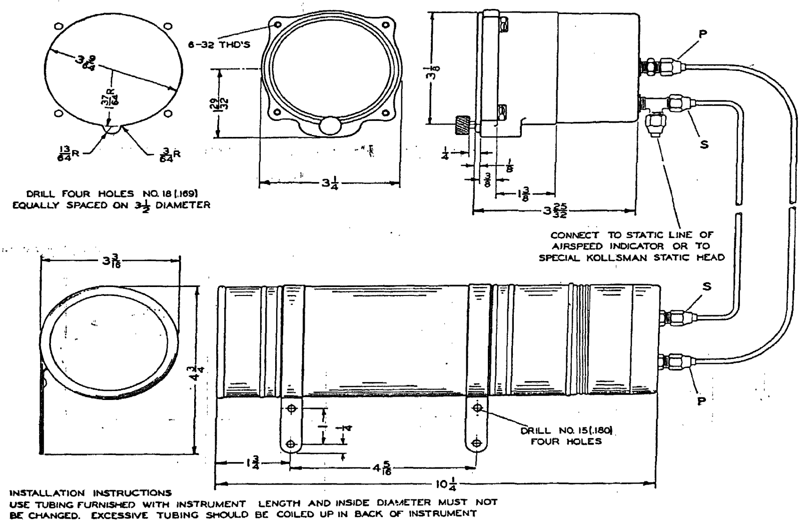
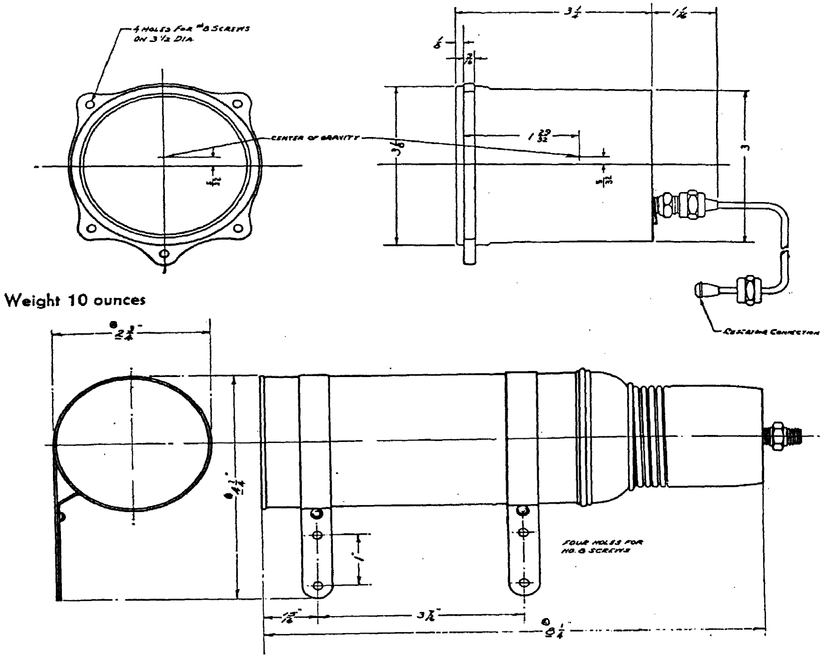


FIGURE 125. KOLLSMAN VERTICAL SPEED
(CLIMB) INDICATOR



Weight 10 ounces

Weight 17 ounces

FIGURE 126. PIONEER CLIMB INDICATOR

Type 1042 Type 1045

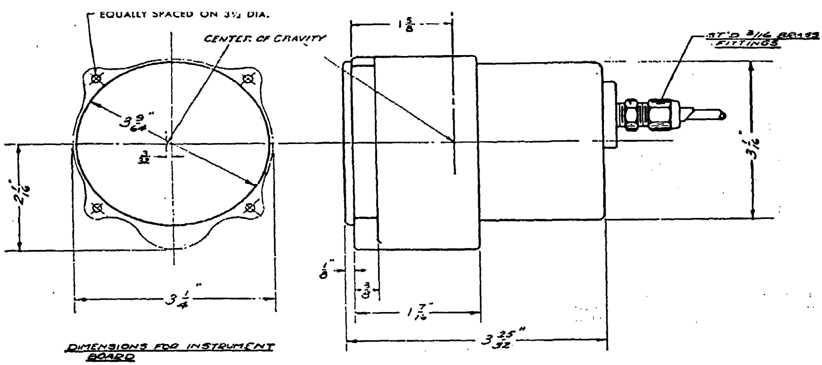


FIGURE 127. PIONEER CLIMB INDICATOR

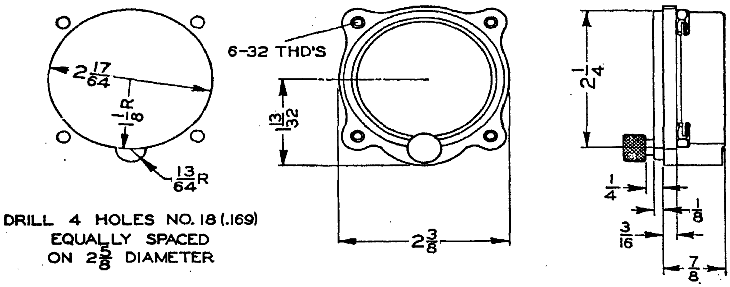


FIGURE 128. KOLLSMAN AVIATION CLOCK

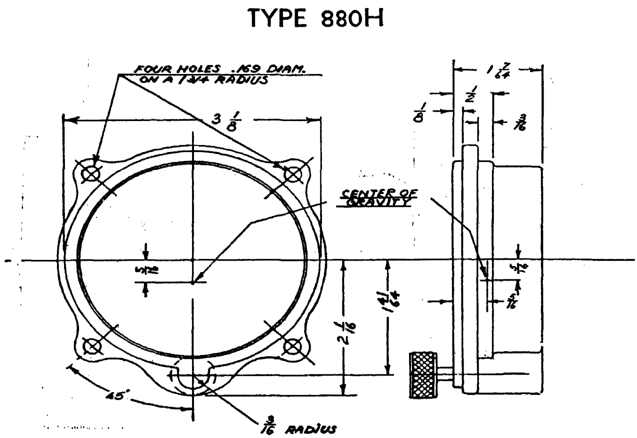


FIGURE 129. PIONEER ELGIN CLOCK

PIONEER-ELGIN CLOCKS
TYPE 757B

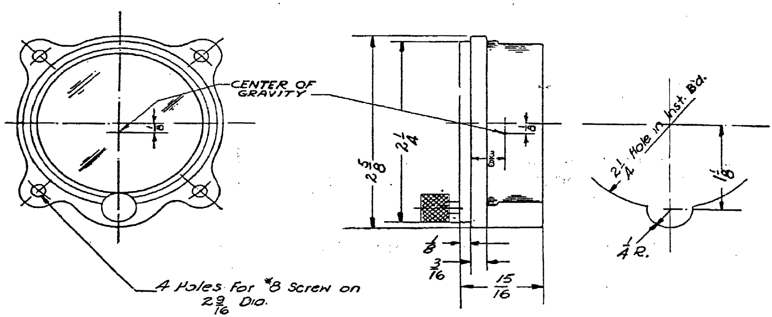


FIGURE 130. PIONEER ELGIN CLOCK

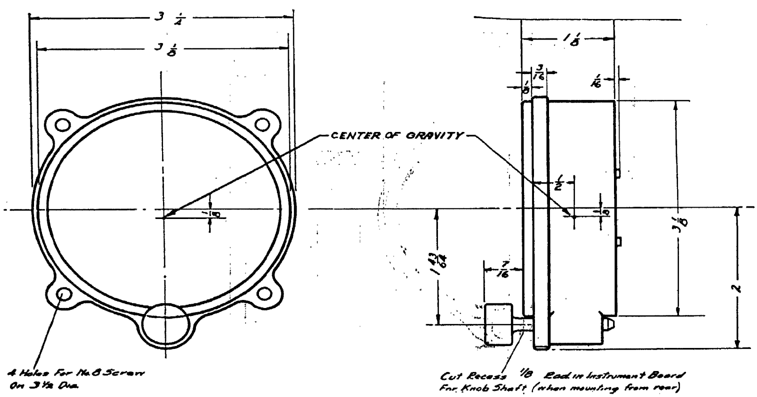


FIGURE 131. PIONEER WALTHAM CLOCK

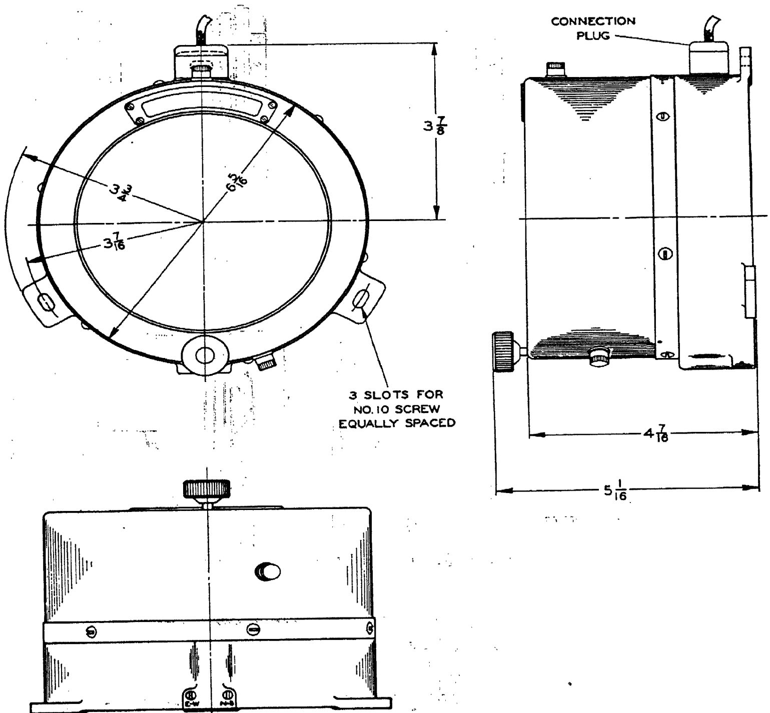


FIGURE 132. KOLLSMAN APERIODIC COMPASS

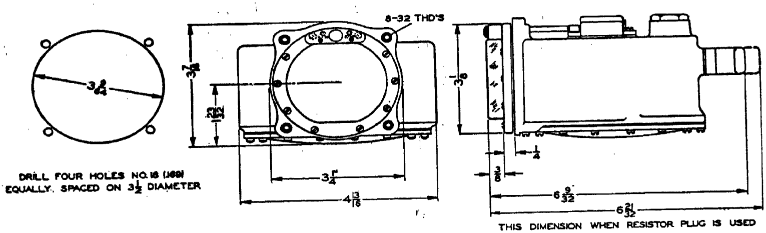


FIGURE 133. KOLLSMAN AVIGATION COMPASS

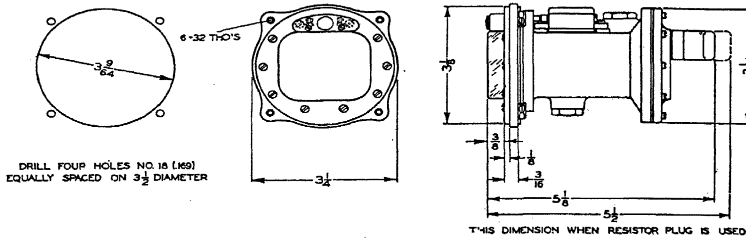


FIGURE 134. KOLLSMAN COMPASS

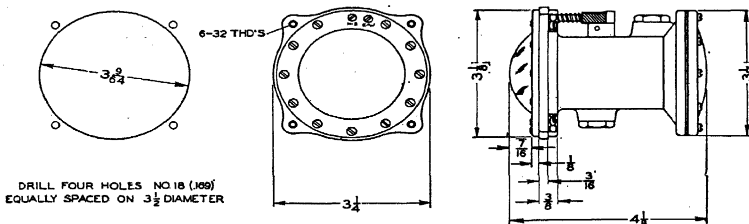
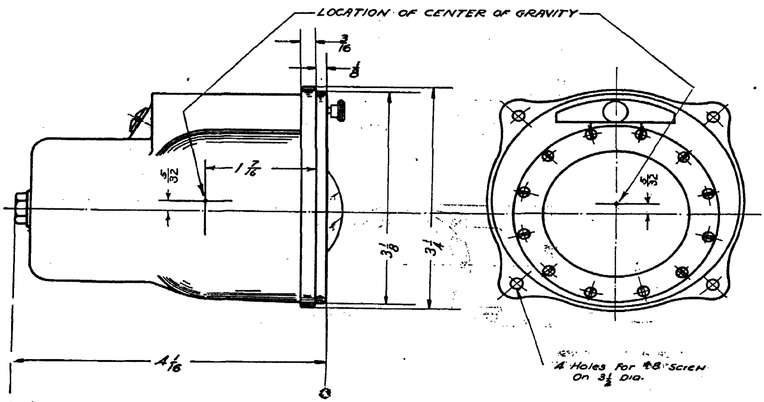
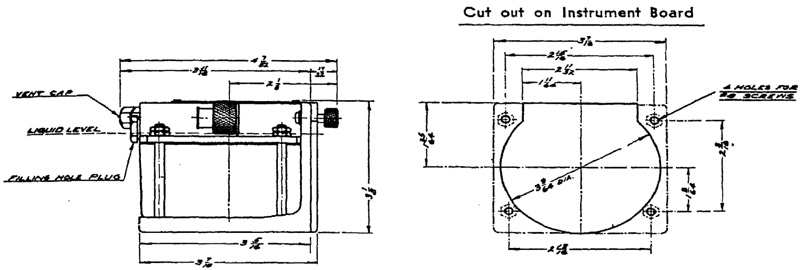


FIGURE 135. KOLLSMAN COMPASS



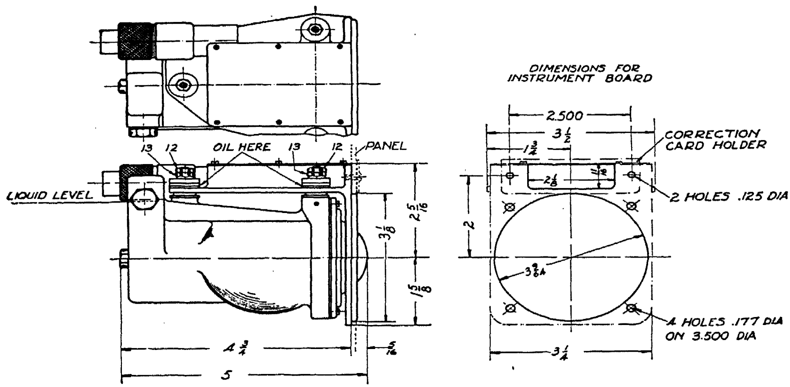
Weight 1.0 lb.

FIGURE 136. PIONEER MAGNETIC TYPE COMPASS



Weight—1 lb. 8 oz.

FIGURE 137. PIONEER MAGNETIC TYPE COMPASS



WEIGHT 2 lb., 5 oz.

FIGURE 138. PIONEER MAGNETIC TYPE COMPASS

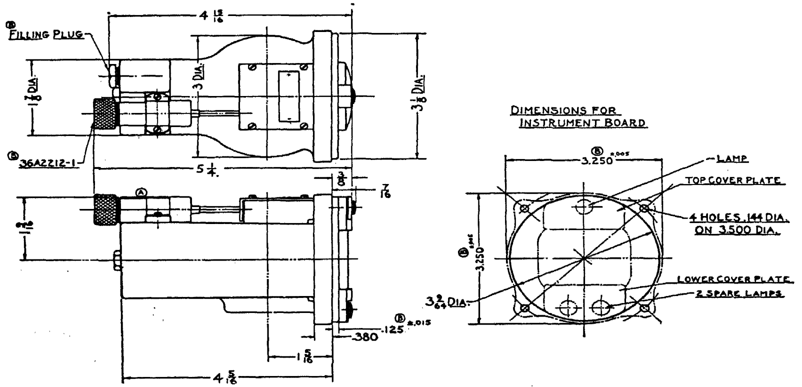


FIGURE 139. PIONEER MAGNETIC TYPE COMPASS

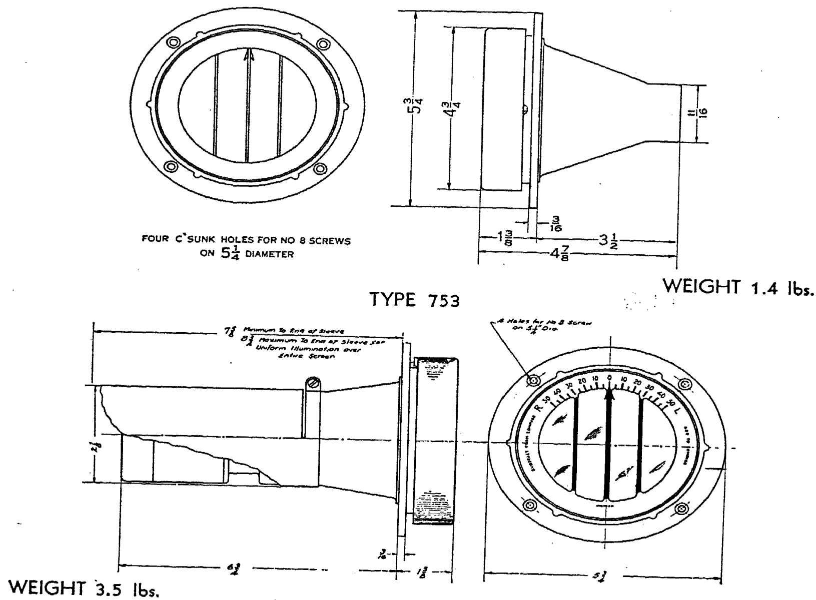


FIGURE 140. PIONEER DRIFT INDICATOR

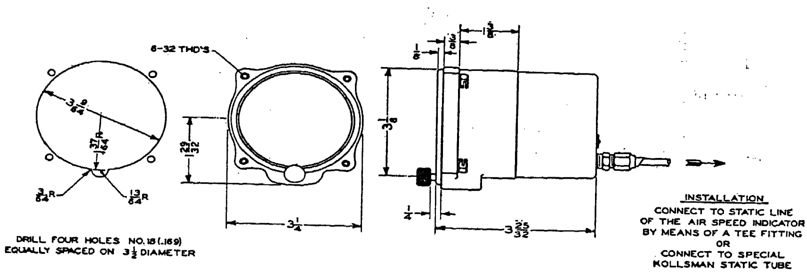


FIGURE 141. KOLLSMAN LEVEL FLIGHT INDICATOR

LANDING GEAR COMPONENTS

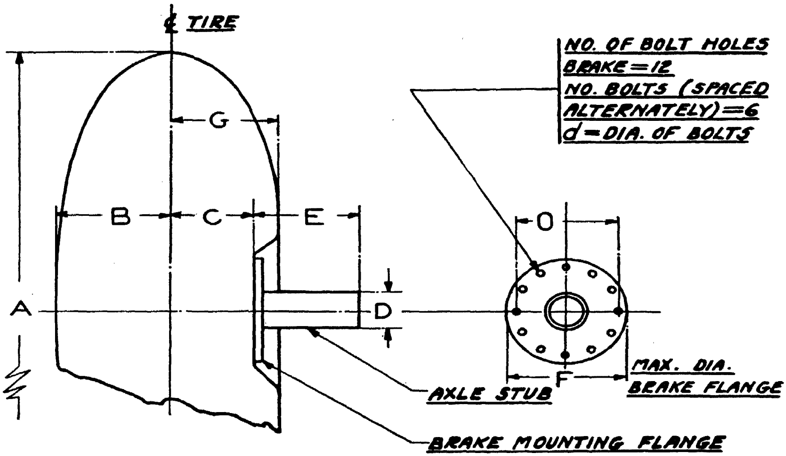


FIGURE 142. SEE TABLE 41 FOR DIMENSIONS OF BENDIX "STREAMLINE" WHEELS

TABLE 41 — *Bendix Cast Streamline Landing Wheels and Brakes*

Size A	B	C	d	D	E	F	G	O	Maximum Weight Pounds		Brake Diam.	Guaranteed Ultimate Strength			
									Wheel & Brake Aluminum Alloy	Stub Axle		Brake Type	Radial Load	Side Load	Max. Tire Load Lb.
*18″	3-7/8	....	*5/16	1.500	3	None	3-43/64	None	7.10	*	4″	Mech.	8,600	1,950	850
21″	3-9/16	2-11/16	1/4	1.500	3	4-11/16	3-7/16	4	15.20	1.90	9″	Mech.	14,000	3,600	1100
24″	4-3/32	3-1/4	1/4	2.000	4	5-9/16	3-15/16	4-3/4	24.25	3.65	10″	Mech.	21,000	5,300	1600
27″	4-19/32	3-5/8	1/4	2.000	4	5-9/16	4-5/16	4-3/4	32.10	4.20	11″	or	25,000	6,350	2200
31″	5-1/4	4-1/16	5/16	2.500	5	6-9/16	4-61/64	5-3/4	40.75	7.25	13″	Hyd.	31,000	8,000	3100
36″	6-1/8	4-5/8	3/8	3.000	6	8-1/8	5-3/4	7	56.85	11.80	15″	Hyd.	37,500	10,500	4600
40″	6-13/16	5	3/8	3.000	6	8-1/8	6-13/32	7	72.25	12.20	17″	Hyd.	50,000	13,250	6000
45″	7-21/32	5-3/4	1/2	3.000	6	9-1/4	7-13/64	8	112.0	14.00	20″	Hyd.	62,500	16,850	8100
50″	8-1/2	6-11/16	1/2	3.750	7½	9-1/4	8	8	120.5	20.85	22″	Hyd.	75,000	20,500	10000
60″	10-3/16	7-3/4	5/8	4.000	8	10-5/8	9-19/32	9	Under	Design	26″	Hyd.	130,000	34,200	17500

* A landing wheel for light planes using 18″ tail wheel streamline tire. Axle furnished by customer. Only one mounting bolt necessary. Refer to Figure 142 for meaning of A, B, C, etc.

TABLE 42 — *Bendix Cast Streamline Tail Wheels*

Size A	*B*	*C*	*Bearing Bore Inboard*	*Bearing Bore Outboard*	*Assembly Weight Alum. Alloy (Lb.)*	*Max. Tire Loading (Lb.)*	*Tire*
8.00″	1-3/4	1-19/32	1.000	.590	1.37	400	40
10.50″	2-9/32	2-5/32	1.250	.625	1.94	700	40
13.25″	2-55/64	2-45/64	1.500	1.000	3.19	1100	40
15.50″	3-11/32	3-5/32	1.750	1.500	4.62	1500	40
18.00″	3-7/8	3-43/64	2.000	1.750	6.46	2000	40
20.00″	4-11/32	4-7/64	2.500	2.000	9.12	2600	42
*28.00″	5-29/32	5-39/64	3.000	2.500	....	5000	42

* Under Design.

Refer to Figure 143.

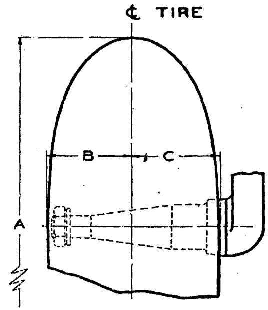


FIGURE 143.
SEE TABLE 42 FOR DIMENSIONS OF BENDIX "STREAMLINE" TAIL WHEELS

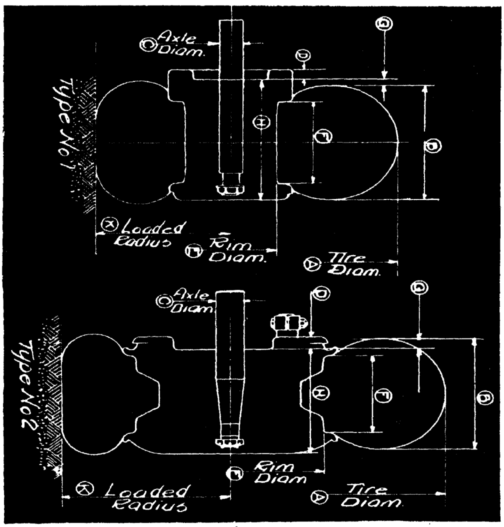


FIGURE 144. SEE TABLE 43 FOR DATA ON GOODRICH LOW PRESSURE TIRES

TABLE 43—*Goodrich Low Pressure Tires—Autofan Wheels*

Tire Size	Wheel Size	Dimensions in Inches								Max. Load Recom-mended Lb.	Inflated Air Pressure Lb.	Loaded Radius Inches K	Type No.	Std. Bolt Circle Dia. Inches	No. and Dia. of Bolt Holes
		A	B	C	D	E	F	G	H						
7.00-5	7.00-5	17.27	6.80	1.312 1.313	.64	5.00	5	.32	7.40	800	15	6.98	1	3	4-17/64″
8.00-5	7.00-5	19.26	7.42					.01		900	15	7.58			
6.50-10	6.50-10	22.03	6.54	1.500 1.499	.78	10.00	$4\frac{3}{4}$	.57	6.43	1300	25	9.25	2	4-3/4	6-25/64″
*7.50-10	6.50-10	23.33	7.09					.83		1600	25	9.66			
8.50-10	8.50-10	25.62	8.67	2.000 1.999	.78	10.00	$6\frac{1}{4}$	.86	7.84	1950	25	10.45	2	4-3/4	6-25/64″
9.50-12	9.50-12	29.12	9.74	2.000 1.999	1.03	12.00	7	1.10	8.46	2600	25	12.0	2	4-3/4	12-25/64″
11.00-12	9.50-12	31.91	10.55					1.50		3400	25	12.83			
11.00-12	11.00-12	31.80	11.07	2.500 2.499	1.03	12.00	$8\frac{1}{4}$	.75	10.35	3400	25	12.78	2	4-3/4	12-25/64″
12.50-14	12.50-14	37.00	12.52	2.500 2.499	1.13	14.00	$9\frac{1}{2}$	1.13	11.37	4700	25	14.88	2	5-3/4	12-25/64″
15.00-16	15.00-16	42.42	14.65	3.000 2.999	1.25	16.00	$11\frac{1}{4}$	1.65	13.25	7000	28	17.04	2	8	12-37/64″

Wheel for this tire not yet available. Tire is mounted on wheel of next smaller size.

Refer to Figure 144.

TABLE 44

Data on Autofan Wheels with Mechanical or Hydraulic Brakes

Wheel Size	*Width of Rim Bet. Flanges*	*Type of Rim*	*Bearings in Wheel*	*Diameter of Axle*	*Maximum Approved Static Load*	*Weight of Alum. Alloy*	*Wheel & Brake Magnesium Alloy*	*Stub Axle Wght. Lb.*
7.00-5	5.0	Split	Timken	1.312	1000	8.75	7.5	2.25
6.50-10	4.75	D. C.	Timken	1.5	1900	15.75	13.0	2.25
8.50-10	6.25	D. C.	Timken	2.0	2000	17.75		3.50
9.50-12	7.0	D. C.	Timken	2.0	3000	25.5	20.0	4.75
11.00-12	8.25	D. C.	Timken	2.5	4050	30.0	24.0	9.0
12.50-14	9.5	D. C.	Timken	2.5	Al. 7200 Mag. 4650	42.5	31.0	9.75
15.00-16	11.25	D. C.	Timken	3.0	7000			

Wheel data not guaranteed.
Autofan Wheels manufactured by Hayes Industries, Inc., Jackson, Mich.

Wheels and wires are manufactured by several companies. The above are given only as typical examples.

Typical design data for shock absorber struts are given in Table 45. The letters A, B, C, etc. refer to the indicated dimensions shown in Figure 145.

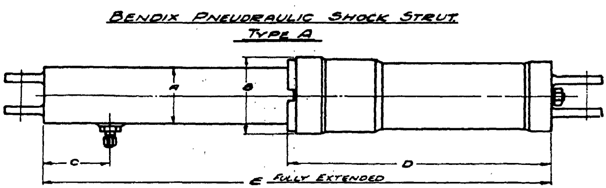


FIGURE 145. FOR GUIDING DIMENSIONS
SEE TABLE 45

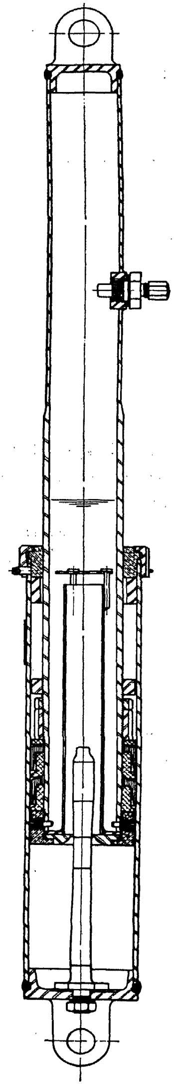


FIGURE 146. CROSS SECTION OF A BENDIX SHOCK ABSORBER STRUT

TABLE 45 — *Specifications*

Static Load per Strut (Lb.)	*Stroke (In.)*	*A*	*B*	*C*	*D*	*E*	*Weight for Length (Lb.)*	*Weight Increment per ½ In. Stroke (Lb.)*	*Length Increment per ½ In. Stroke (In.)*
500—1000	6	$1\frac{1}{2}$	$2\frac{3}{8}$	$2\frac{9}{16}$	10	$19\frac{7}{16}$	$5\frac{3}{4}$	$\frac{1}{4}$	1
1000—1500	6	$1\frac{3}{4}$	$2\frac{5}{8}$	$2\frac{1}{2}$	10	$19\frac{3}{8}$	$6\frac{3}{4}$	$\frac{1}{4}$	1
1500—2000	6	2	$2\frac{7}{8}$	$2\frac{7}{16}$	10	$19\frac{5}{16}$	$7\frac{3}{4}$	$\frac{5}{16}$	1
2000—2500	6	$2\frac{1}{4}$	$3\frac{1}{8}$	$2\frac{3}{8}$	10	$19\frac{1}{4}$	9	$\frac{3}{8}$	1
2500—3000	6	$2\frac{1}{2}$	$3\frac{3}{8}$	$2\frac{3}{8}$	$10\frac{1}{8}$	$19\frac{3}{8}$	$10\frac{1}{2}$	$\frac{7}{16}$	1
3000—4000	6	$2\frac{3}{4}$	$3\frac{5}{8}$	$2\frac{3}{8}$	$10\frac{1}{8}$	$19\frac{3}{8}$	$12\frac{1}{4}$	$\frac{1}{2}$	1
4000—5000	6	$3\frac{1}{8}$	$4\frac{3}{16}$	$2\frac{3}{8}$	$10\frac{3}{4}$	$20\frac{1}{8}$	16	$\frac{5}{8}$	1
5000—6000	6	$3\frac{1}{2}$	$4\frac{9}{16}$	$2\frac{5}{16}$	$10\frac{3}{4}$	$20\frac{1}{8}$	$19\frac{1}{4}$	$\frac{3}{4}$	1
6000—8000	6	$4\frac{1}{8}$	$5\frac{3}{16}$	$2\frac{5}{16}$	$10\frac{3}{4}$	$20\frac{1}{8}$	$24\frac{3}{4}$	1	1
8000—10,000	6	$4\frac{5}{8}$	$5\frac{13}{16}$	$2\frac{5}{16}$	$11\frac{5}{8}$	$20\frac{13}{16}$	$31\frac{3}{4}$	$1\frac{1}{4}$	1
10,000—12,000	6	5	$6\frac{3}{16}$	$2\frac{5}{16}$	$11\frac{5}{8}$	$20\frac{13}{16}$	$35\frac{1}{4}$	$1\frac{3}{8}$	1

Dimension C is located with the taxiing position at 3 inches from the end of stroke and a load factor of 3 due to compression of air only at the end of stroke.

The last two columns give data for determining weights and lengths for other strokes.

Weights and lengths do not include end fitting lugs which must be designed to suit conditions.

POWER PLANT ACCESSORIES

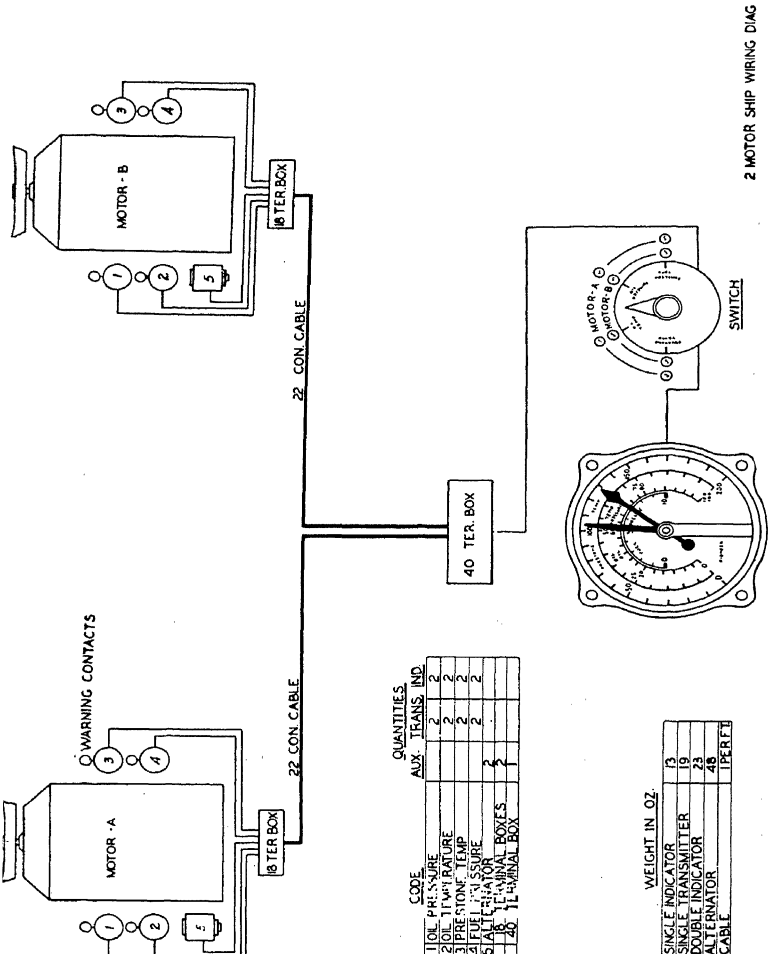


FIGURE 147. TWO-ENGINE AIRPLANE WIRING DIAGRAM FOR PIONEER AUTOSYN

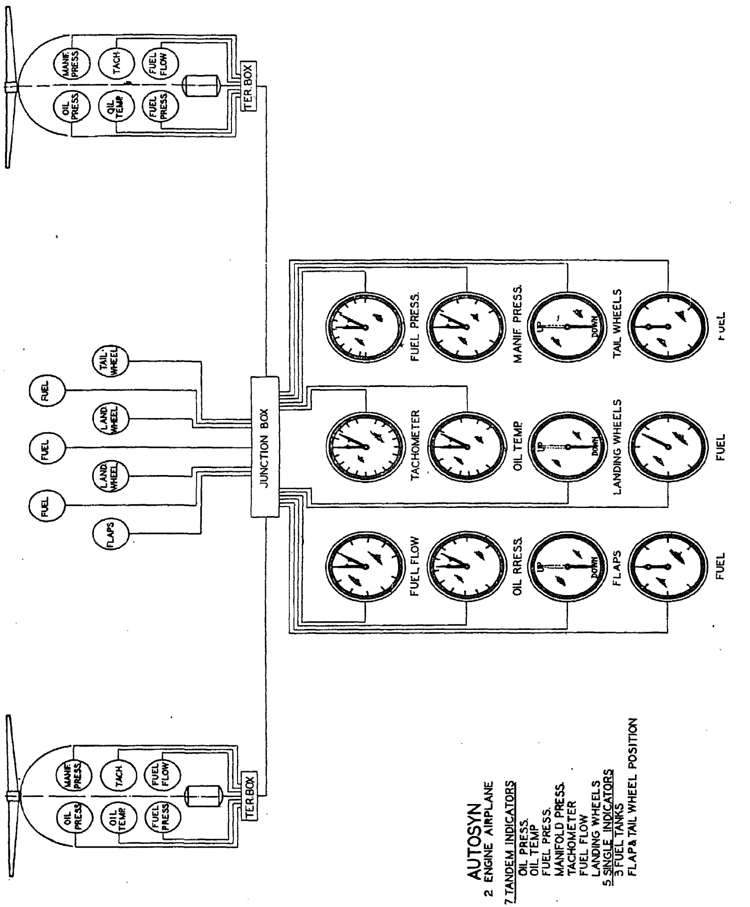


FIGURE 147 A. AUTOSYN ARRANGEMENT FOR A TWO-ENGINE AIRPLANE

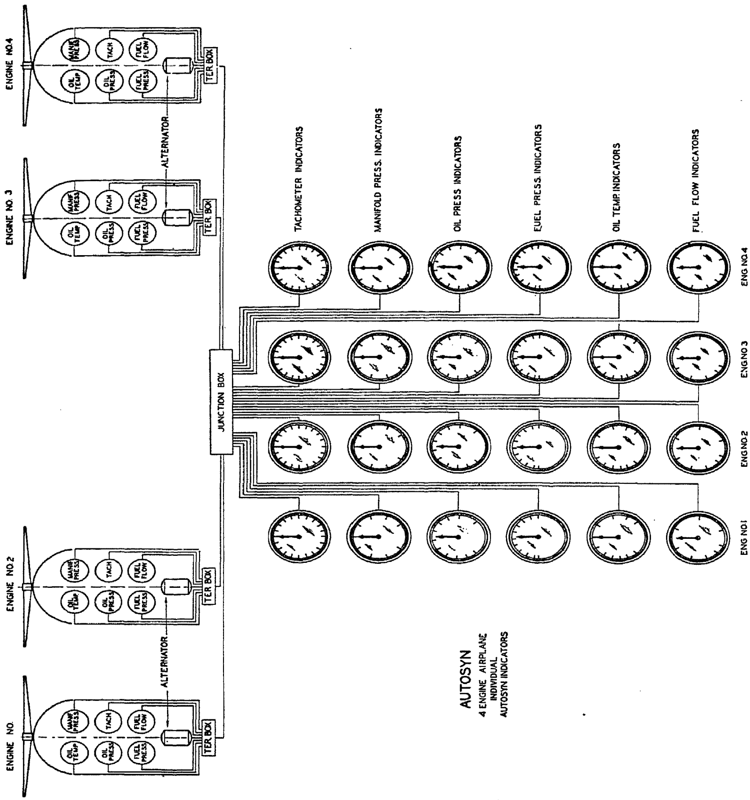


FIGURE 147 B. AUTOSYN ARRANGEMENT FOR A FOUR-ENGINE AIRPLANE: INDIVIDUAL INDICATORS

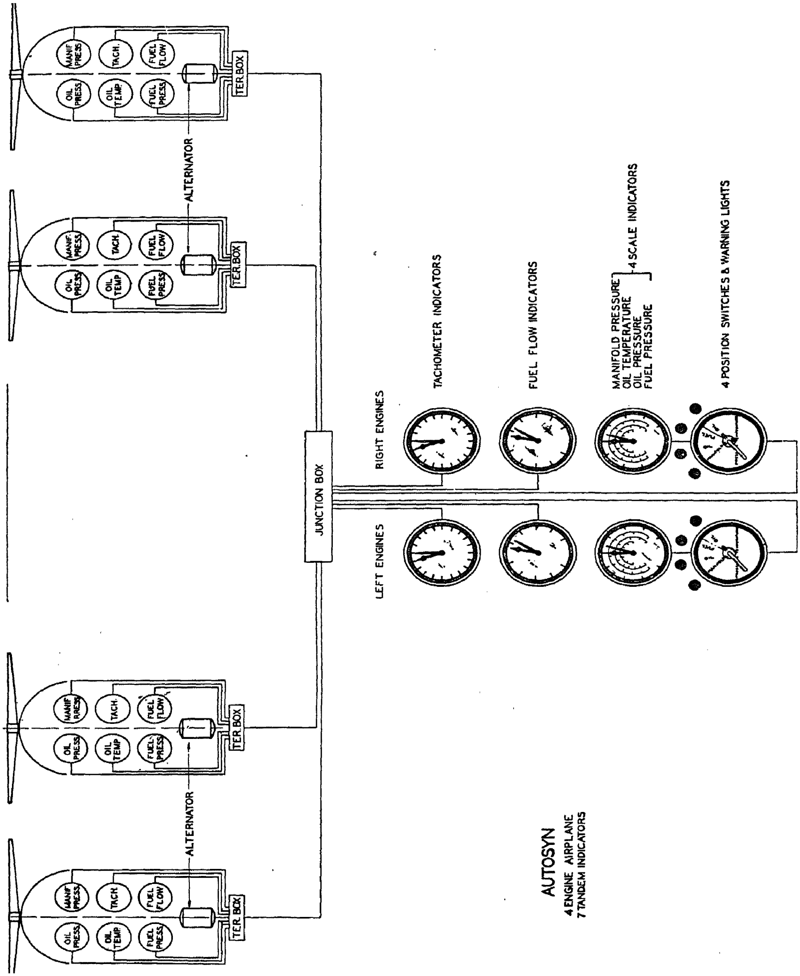
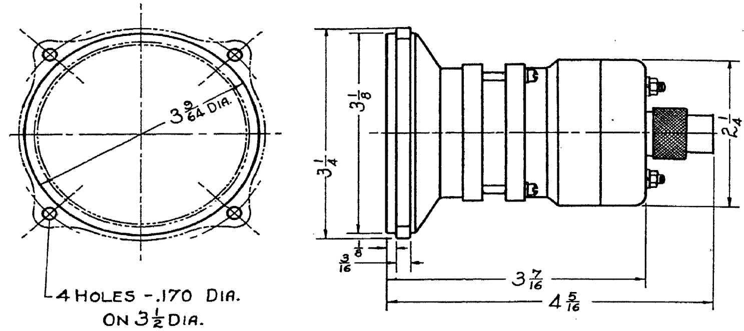


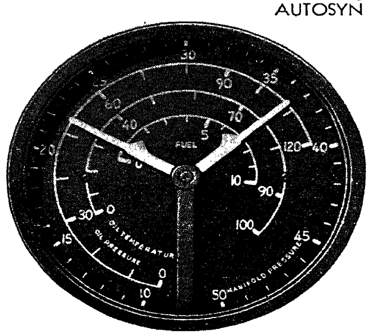
FIGURE 147 C. AUTOSYN ARRANGEMENT FOR A FOUR-ENGINE AIRPLANE—TANDEM ARRANGEMENTS

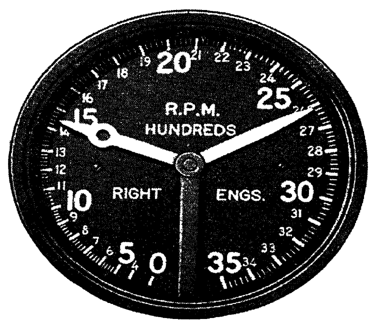
INSTALLATION AND WIRING



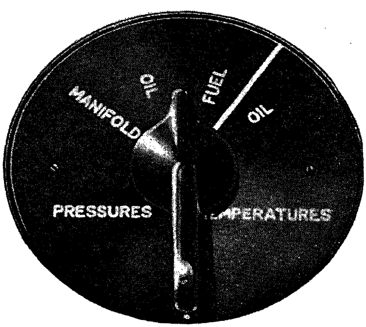
Mounting Dimensions
AUTOSYN INDICATOR

FIGURE 148



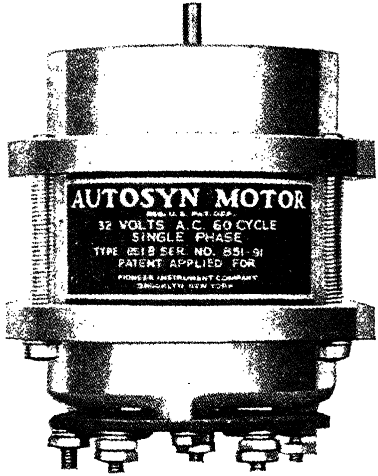


AUTOSYN DUAL INDICATOR



FOUR-SCALE DUAL INDICATOR
and
SELECTOR SWITCH

FIGURE 149



TYPE 851B

FIGURE 150

TABLE 46

Autosyn Specifications

Size	*Type 769 C* 2⅜ in. Dia. x 2 27/32 long	*Type 851 B* 2⅜ in. Dia. x 3 3/32
Weight	10 ounces	15.3 ounces
Current	.16 amperes	.22 amperes
Voltage	32 volt AC 60 cycles	32 Volt AC
Power	2 Watts	3 Watts
Peak Orque	40 Gram—Centimeter	120 Gram—Centimeter
Max Lag	$1\frac{1}{2}$ degrees (Approx.)	$1\frac{1}{2}$ degrees (Approx.)
Shaft diameter	$\frac{1}{16}$ inch	$\frac{1}{8}$ inch

These motors will operate satisfactorily with a line resistance up to 40 ohms.

AUTOSYN MOTORS

TYPE 769C TYPE 851B

TORQUE CURVES

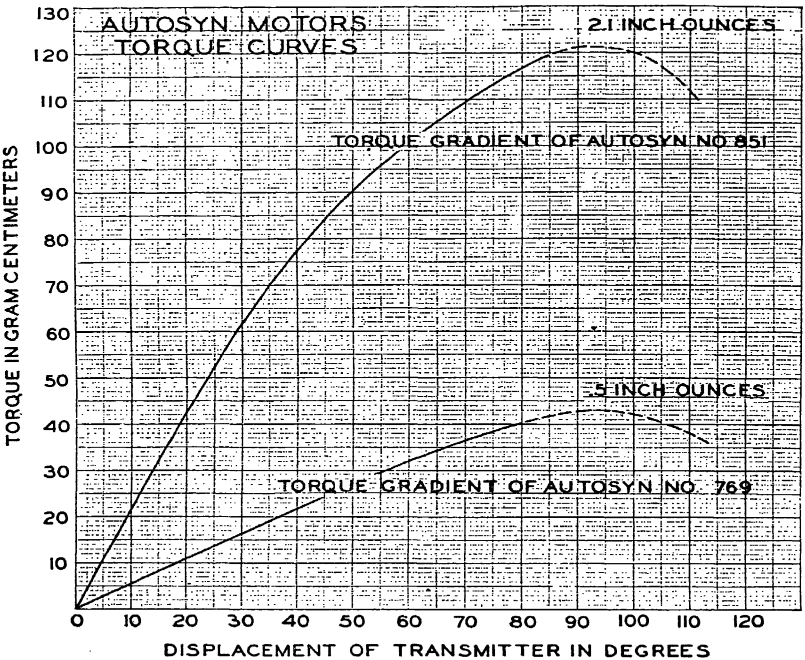


FIGURE 151

AUTOSYN TYPE 769C

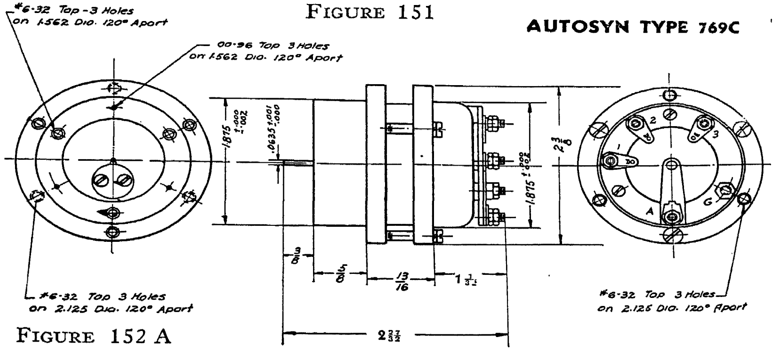


FIGURE 152 A

AUTOSYN 851B

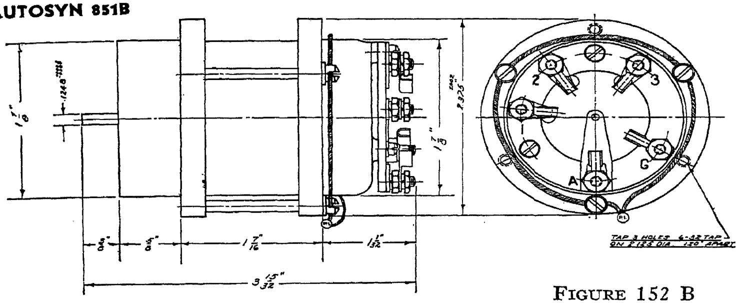


FIGURE 152 B

Instrument Board Cut-out and Installation Dimensions

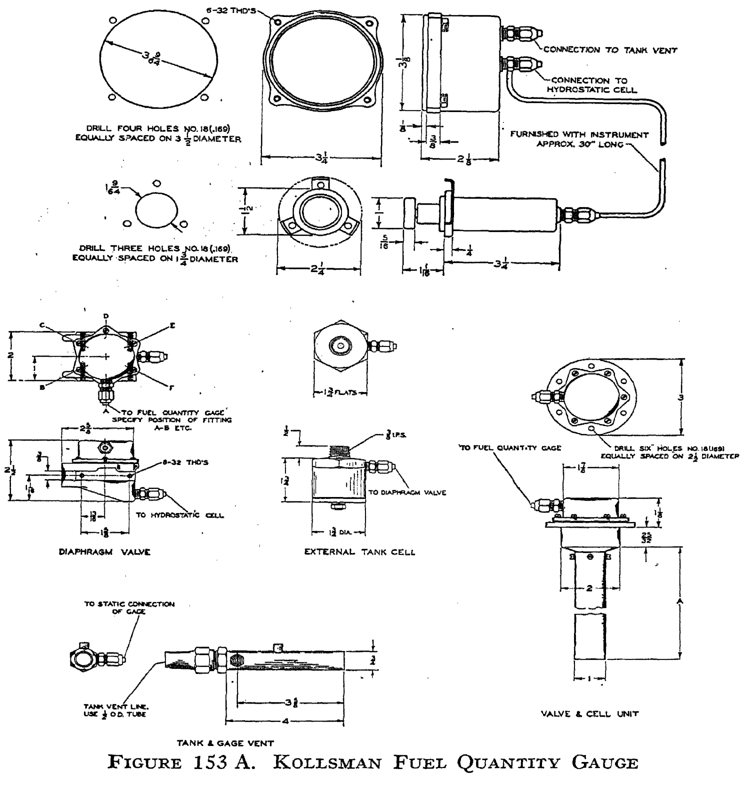
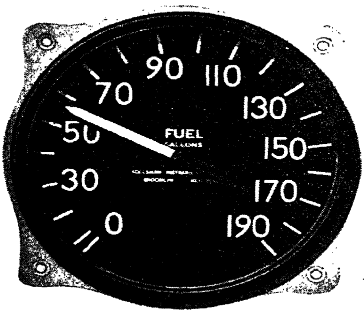


FIGURE 153 A. KOLLSMAN FUEL QUANTITY GAUGE



TYPE 180-01

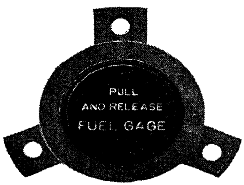


FIGURE 153 B. KOLLSMAN FUEL QUANTITY GAUGE

TYPE 391

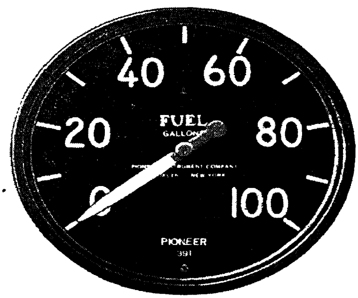
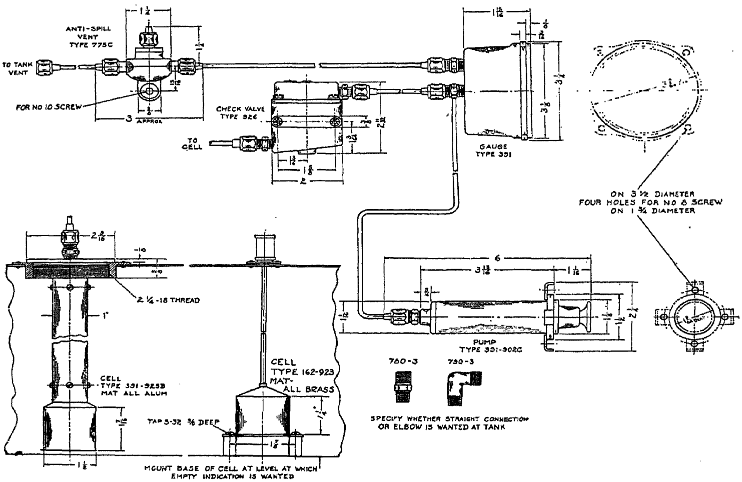


FIGURE 154 B.
PIONEER FUEL LEVEL GAUGE

FUEL LEVEL GAUGE
Type 391



Weights:
Gauge 391, 0.7 lb.
Pump 391-902C, 0.2 lb.
Cell 162-923 with 30″ tube, 0.3 lb.
Cell 391-925B with 30″ tube, 75 lb
Anti Spill Vent 775C, 0.4 lb.
Check Valve 926, 0.5 lb

FIGURE 154 A. PIONEER FUEL LEVEL GAUGE

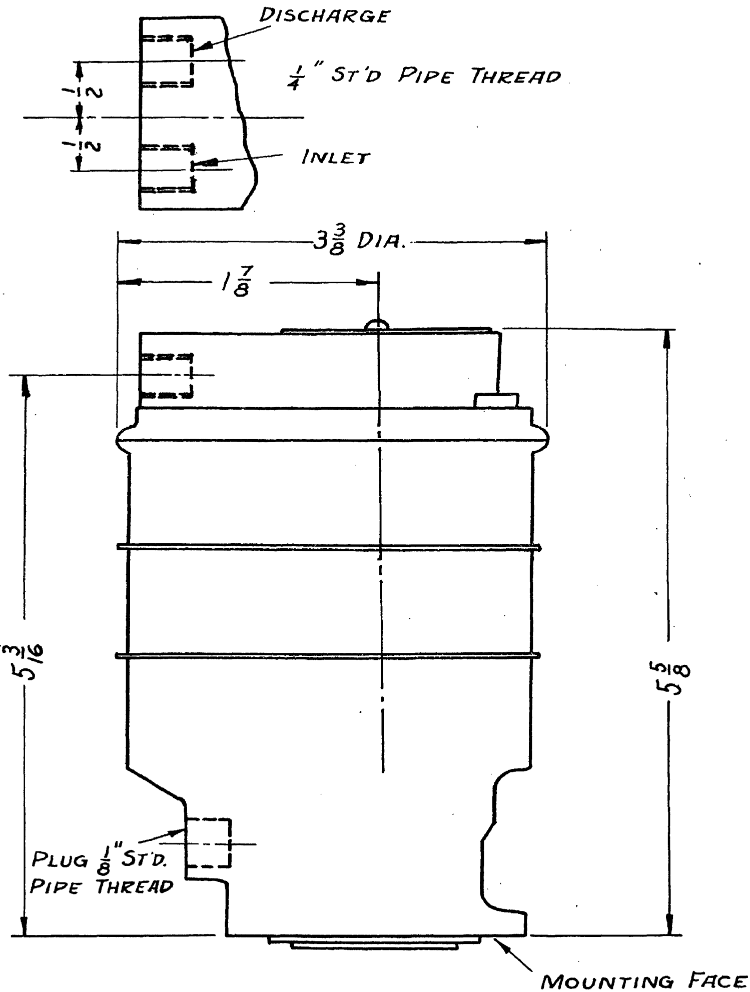


FIGURE 155. ECLIPSE GENERATOR: SEE TABLE 47 FOR CAPACITIES

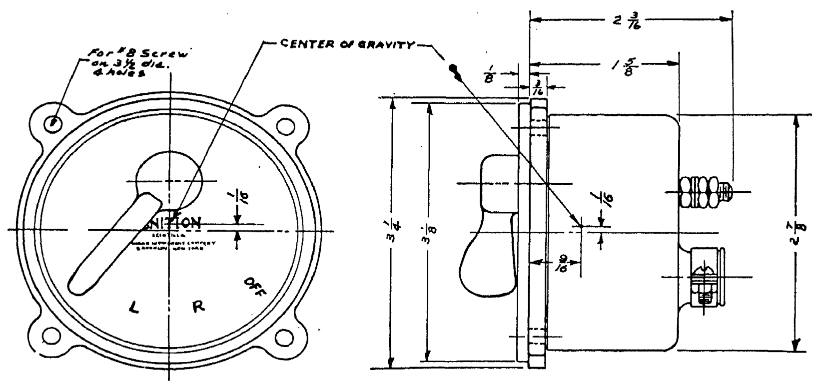
TABLE 47. (Engine Driven Eclipse Generators)

Eclipse generators are available in five types of various capacities as follows:

Type	*Volts*	*Amperes*	*Watts*	*Weight*
G	15	15	225	16 lb.
D	15	25	375	20 lb.
E	15	50	750	29 lb.
D	30	10	300	20 lb.
E	30	20	600	29 lb.

Control Box (permanent mount) for 15 or 30 volt operation $3\frac{1}{2}$ lb.
Control Box (quick detachable) for 15 or 30 volt operation 5 lb.
FB-5 Filter Unit—for 15 or 30 volt operation 4 lb.

NOTE: The above weights include shielded terminal covers which are standard on all generators, control boxes and filter unit. The covers are threaded for the attachment of metallic conduit for radio shielding.



Weight: 9½ oz

FIGURE 156. PIONEER IGNITION SWITCHES

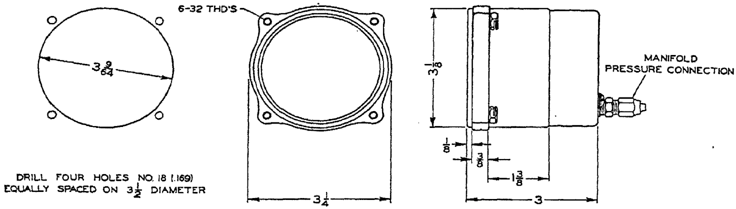


FIGURE 157. KOLLSMAN MANIFOLD PRESSURE GAUGE

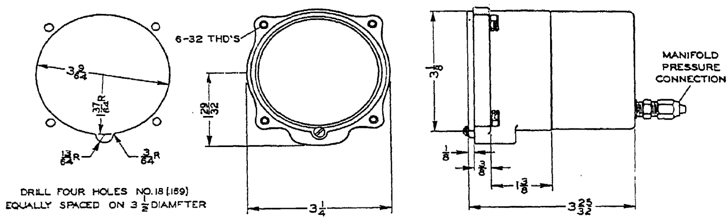


FIGURE 158. KOLLSMAN MANIFOLD PRESSURE GAUGE

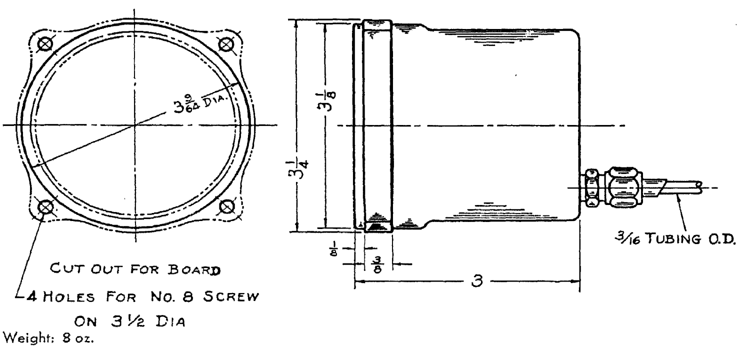
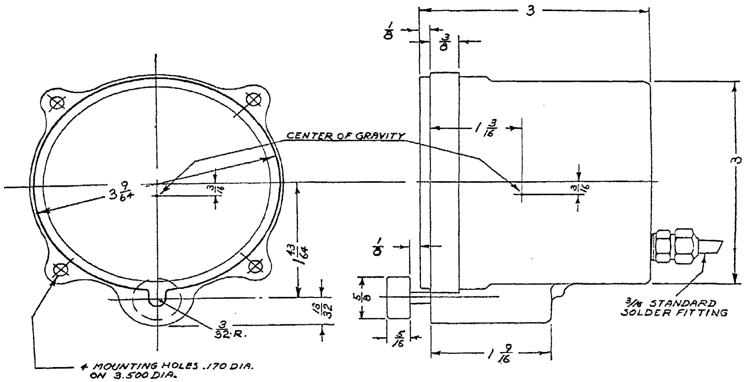


FIGURE 159. PIONEER MANIFOLD PRESSURE GAUGE



Weight: 15 oz.

FIGURE 160. PIONEER MANIFOLD PRESSURE GAUGE

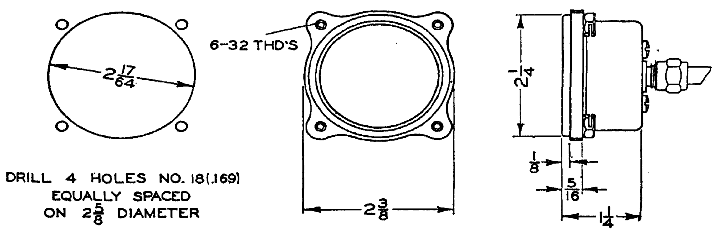


FIGURE 161. KOLLSMAN PRESSURE AND SUCTION GAUGE

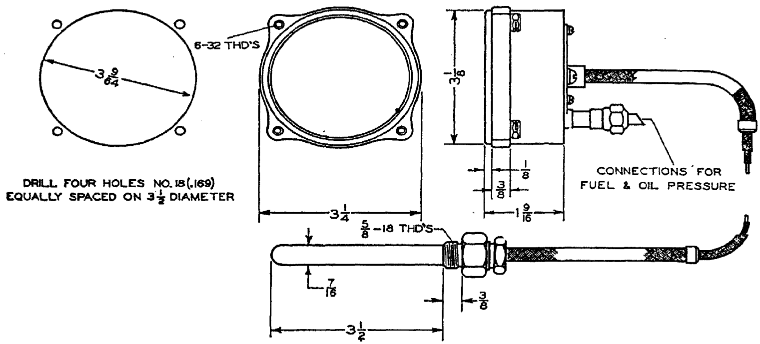


FIGURE 162. KOLLSMAN ENGINE GAUGE UNIT

Registers oil temperature, fuel and pressures, or oil temperature and pressure.

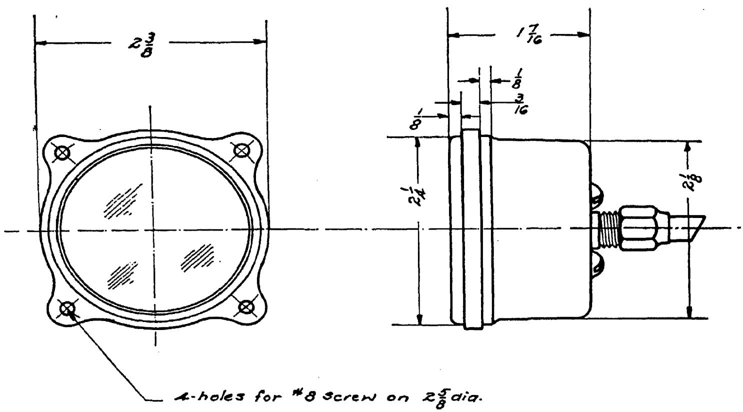
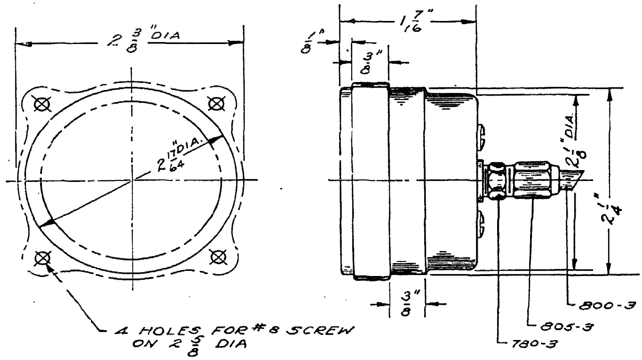


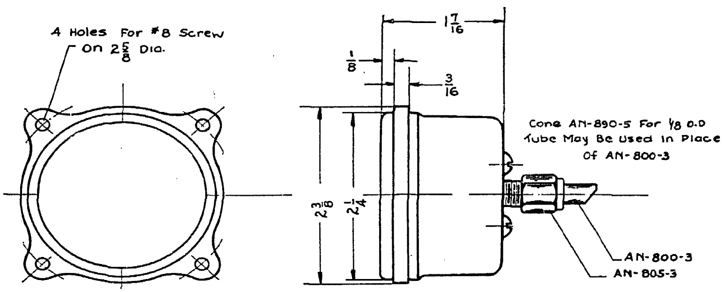
FIGURE 163. PIONEER PRESSURE GAUGE

TYPE 868



Weight: 5½ oz.

TYPE 763



Weight: 4 oz.

FIGURE 164. PIONEER SUCTION GAUGE

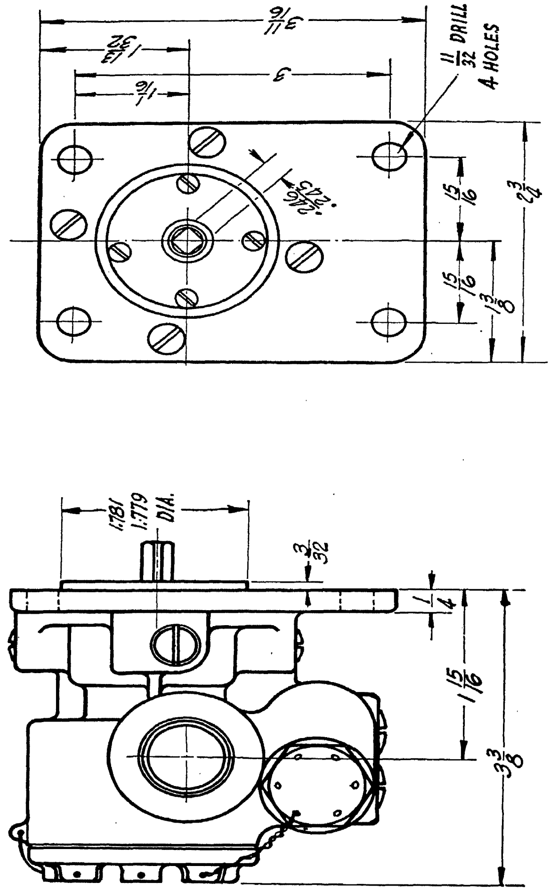


FIGURE 165. EVANS E-7 AIRCRAFT ENGINE FUEL PUMP

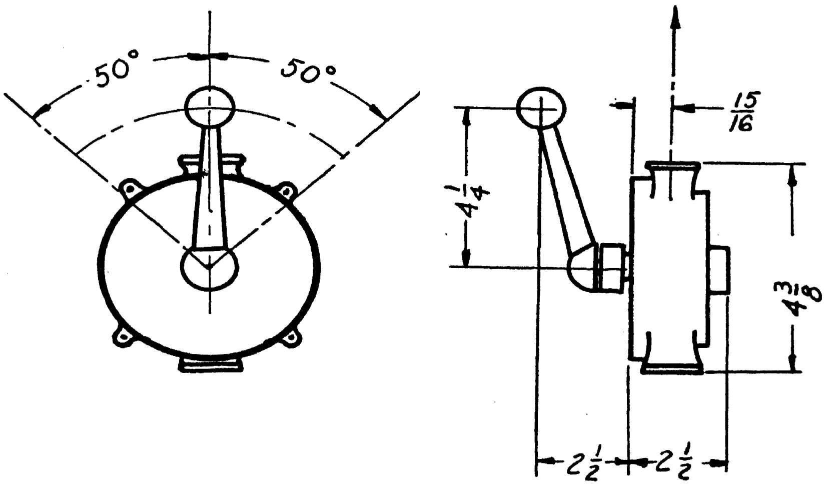
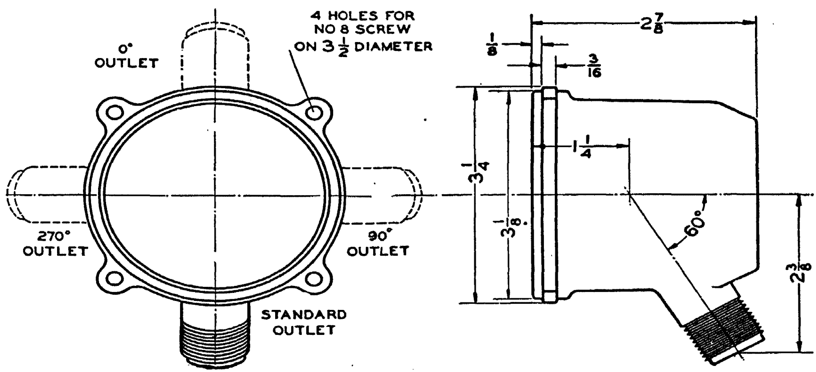


FIGURE 166. HAND OR "WOBBLE" PUMP



WEIGHT 0.9 lbs.

FIGURE 167. PIONEER CENTRIFUGAL TYPE TACHOMETER

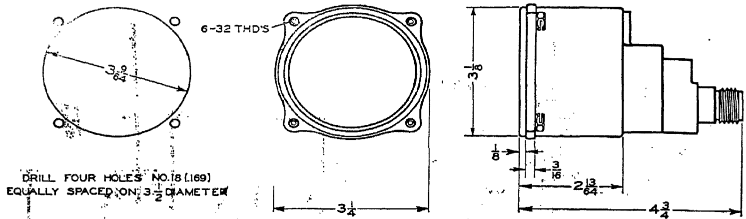


FIGURE 168. KOLLSMAN MAGNETIC TYPE TACHOMETER

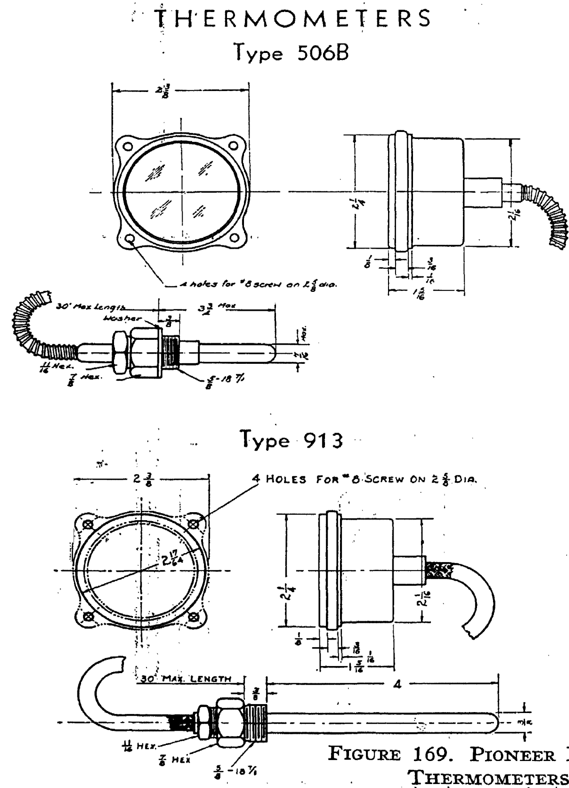


FIGURE 169. PIONEER ENGINE THERMOMETERS

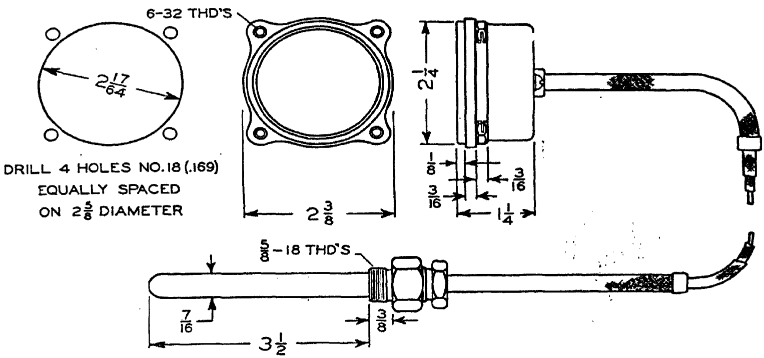


FIGURE 170. PIONEER ENGINE THERMOMETER

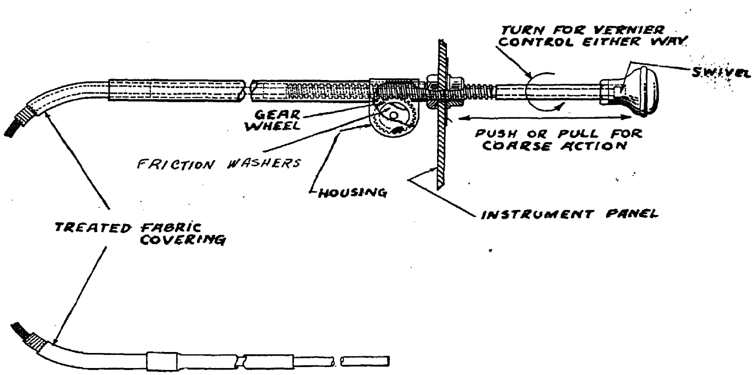


FIGURE 171. ARENS VERNIER CONTROL FOR ENGINES

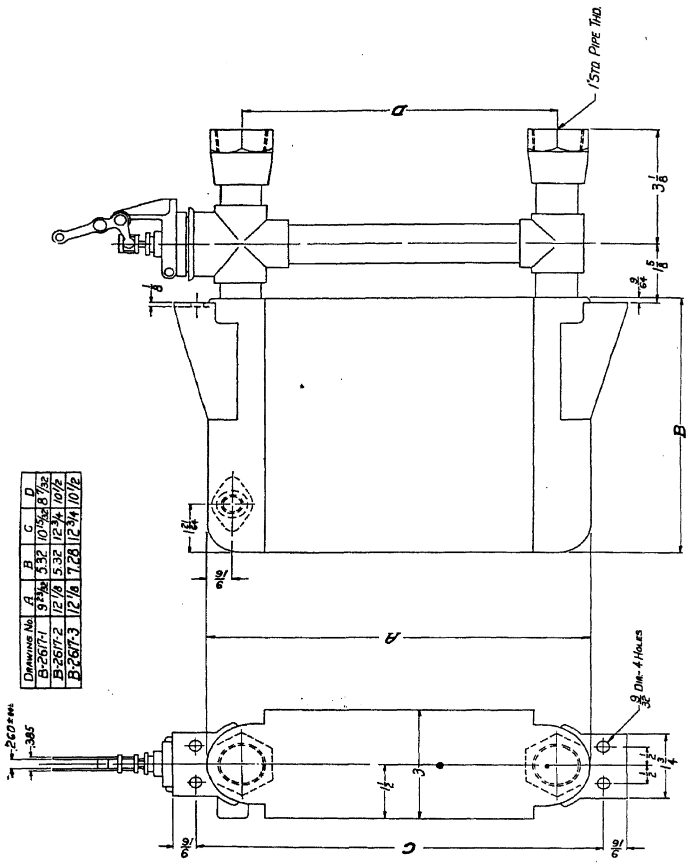


FIGURE 172. G. & O. OIL COOLER

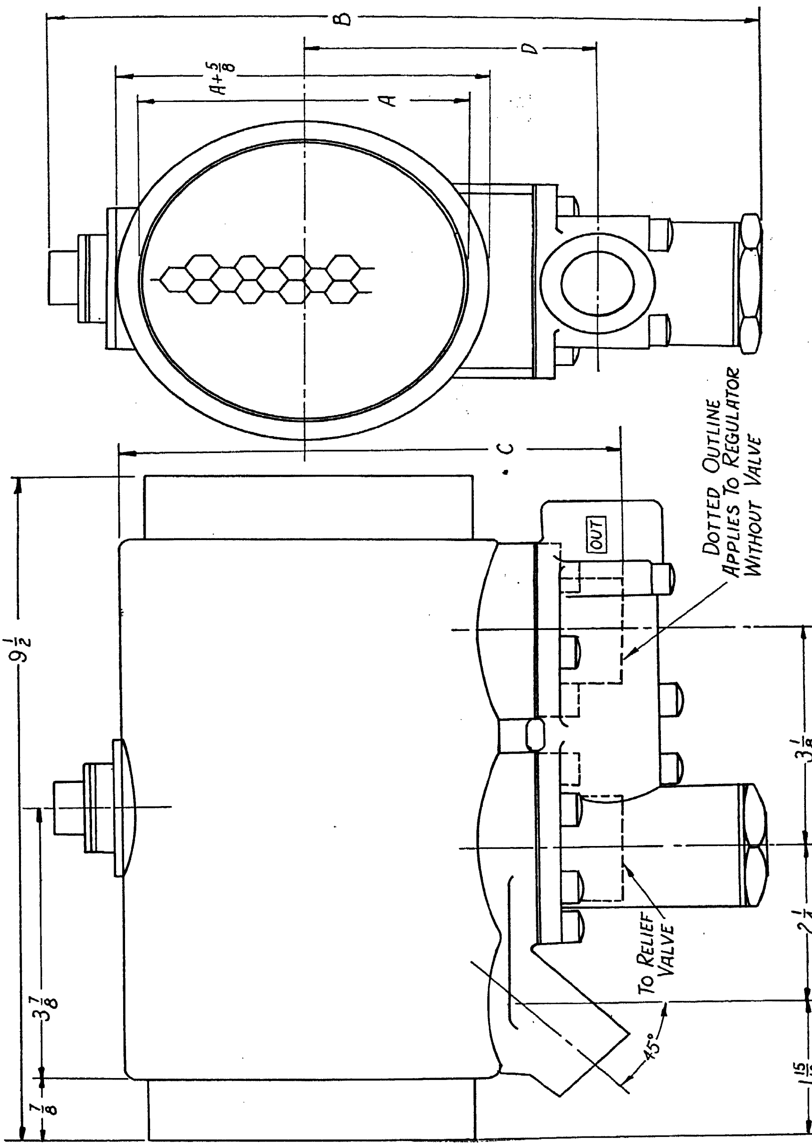


FIGURE 173. OIL COOLER MANUFACTURED BY UNITED AIRCRAFT PRODUCTS, INC.

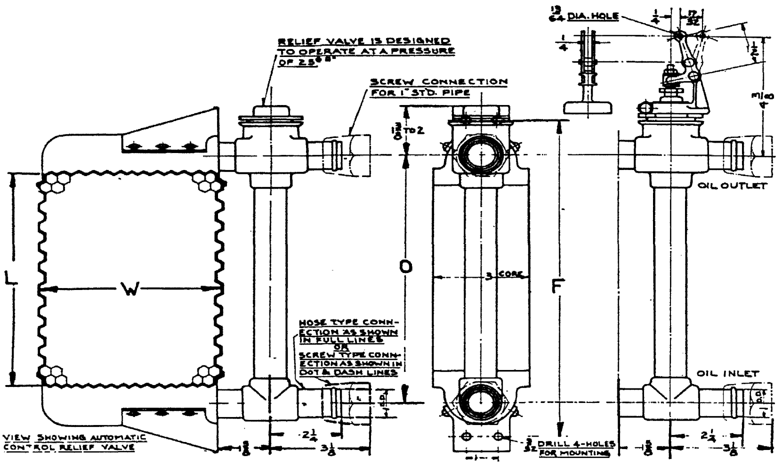


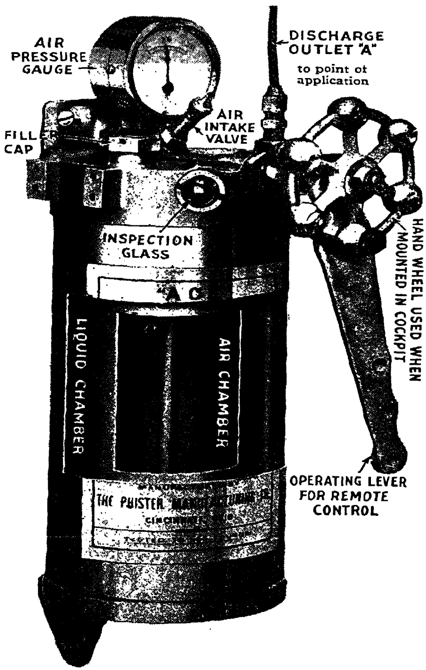
FIGURE 174. OIL COOLER MANUFACTURED BY HARRISON RADIATOR CORP.

TABLE 48

Symbol No.	Part No.	B/P Number	No. Tubes	Recommended Horsepower	Airswept Surface Sq. In.	Oilswept Surface Sq. In.	L	W	O	F	Control	Approx. Weight
1	3074525	C-2308	199	100	428	351	4.812	2.6	6-1/2	8-13/16	Aut.	4.50
2	3074410	C-2302	409	100-350	880	720	4.812	5.2	5-15/16	8-3/16	Aut.	6.65
3	3074475	C-2297	604	350-450	1300	1064	7.031	5.2	8-7/32	10-13/32	Aut.	8.00
4	3074510	C-2307	799	450-550	1720	1407	9.312	5.2	10-1/2	12-3/4	Aut.	9.50
5	3074325	C-2291	799	450-550	1720	1407	9.312	5.2	10-1/2	12-3/4	Aut.	10.10
6	3074275	C-2280	1127	550 up	2426	1985	9.039	7.28	10-1/2	12-3/4	Aut.	12.10
7	3074310	C-2347	1127	550 up	2426	1985	9.039	7.28	10-1/2	12-3/4	Man.& Aut.	12.30

Harrison Radiator Corp., Lockport, N. Y., Outline Dwg. No. D-1430, Aviation Oil Cooler

MISCELLANEOUS EQUIPMENT



Sectional View of the Ace
(Patents Pending)

FIGURE 175. FIRE EXTINGUISHER

Overall height	12½ inches
Greatest width	7 inches
Weight filled	10½ pounds

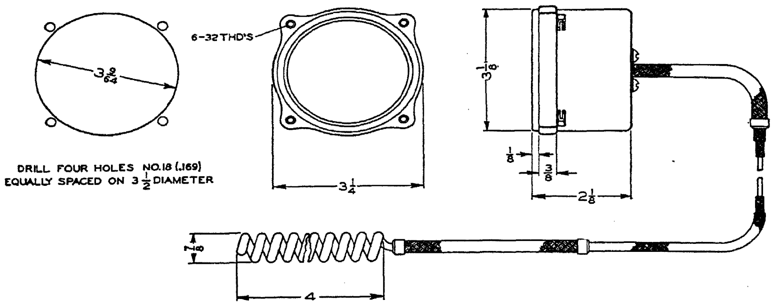


FIGURE 176. KOLLSMAN ICE WARNING INDICATOR

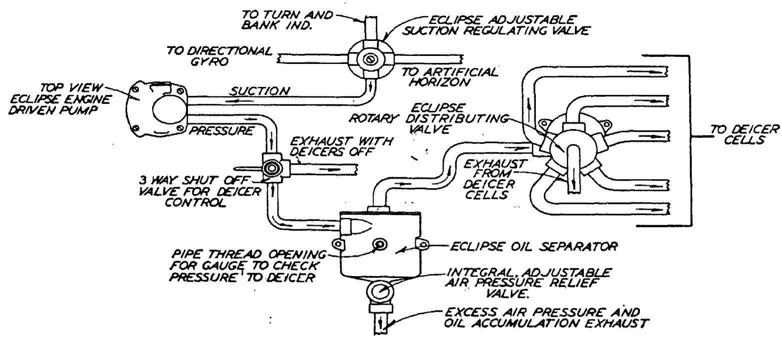


FIGURE 177. TYPICAL ECLIPSE DE-ICER INSTALLATION

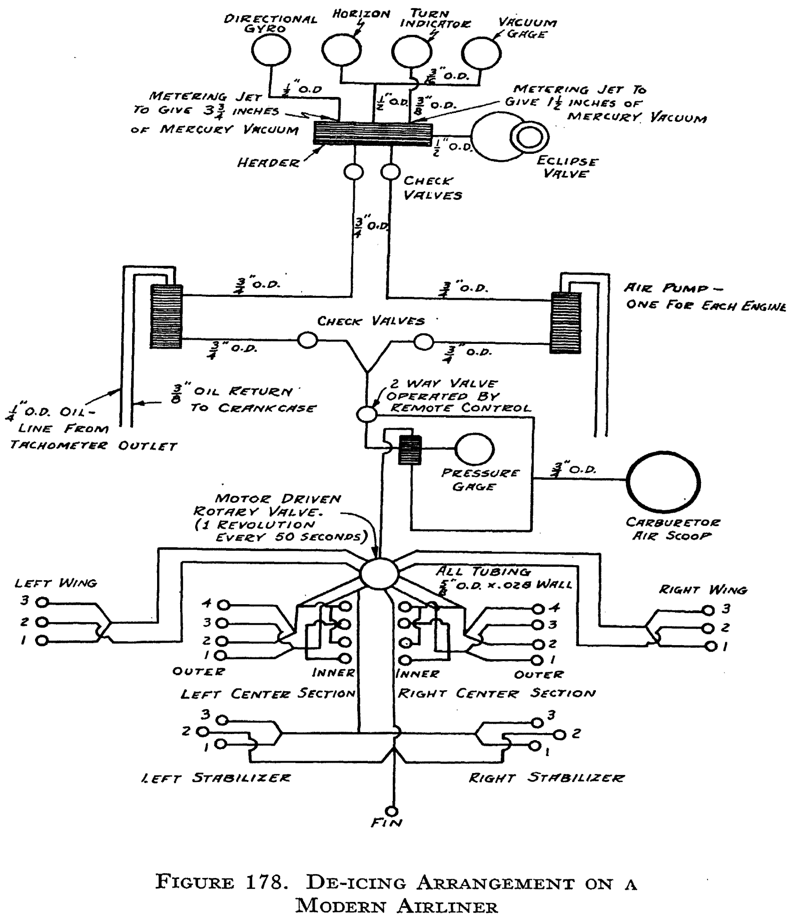


FIGURE 178. DE-ICING ARRANGEMENT ON A MODERN AIRLINER

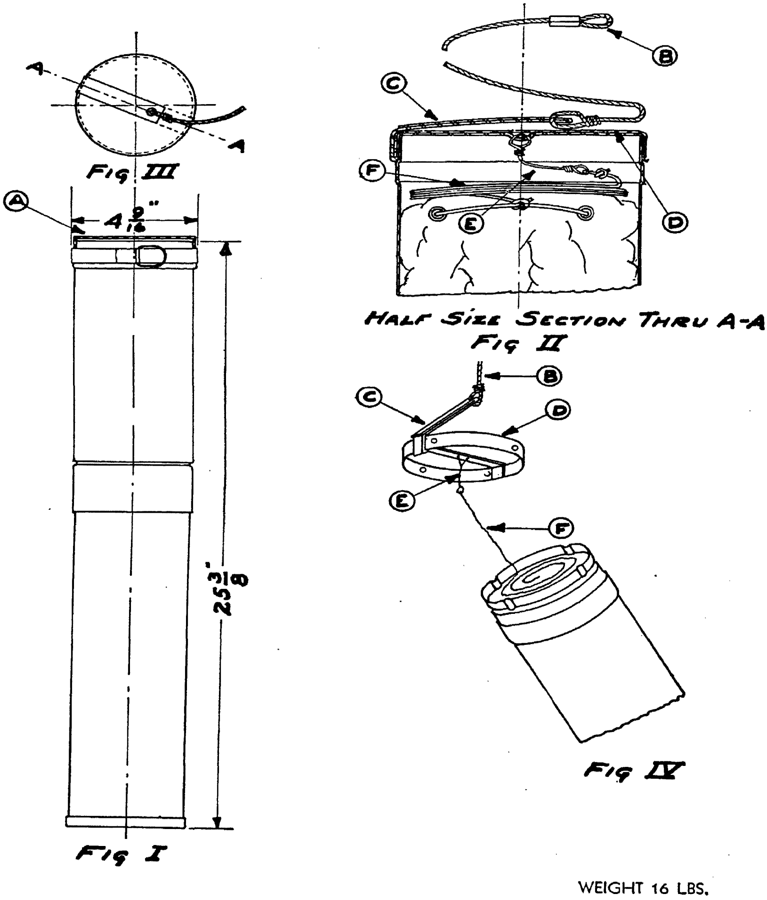


FIGURE 179. PIONEER PARACHUTE FLARE

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